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A STUDY OF PERSONNEL PROPULSION DEVICES FOR USE IN THE VICINITY OF THE MOON

VOLUME I

By N. Economou et al.

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FOREWORD

This report describes the results of a study effort on Personnel Propulsion Devices for Use in the Vicinity of the Moon conducted for the National Aeronautics and Space Administration, Langley Research Center, Hampton, Virginia under contract NAS 1-4098.

The study was performed by personnel of the Bell Aerosystems Company under the technical direction of Dr. L. M. Seale Chief, Space Systems Advanced Design and Mr. N. Economou. Mr. Economou served as the project engineer for the program. The following technical personnel, to whom acknowledgement is hereby made, contributed to the preparation of this report:

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ABSTRACT

This report presents results of studies to establish conceptual configurations of propulsion devices which can be used for transportation on the moon, and for escape from the surface of the moon and injection into the lunar orbit. The study was directed toward "simple" devices which make maximum utilization of the perceptual and control abilities of the pilot and minimum automatic flight control and guidance equipment. The scope of the program was to provide design data and vehicle dynamic characteristics in parametric form to permit the NASA, through simulation studies, to evaluate, refine and select an optimum or near optimum configuration for the intended mission.

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I. INTRODUCTION AND SUMMARY

A. INTRODUCTION

The Personnel Propulsion Devices study is a comprehensive investigation of minimum complexity "flying" devices for use in the vicinity of the moon which provide: (1) transportation for moon-based personnel between two or more points on the lunar surface, and (2) escape capability from the lunar surface, followed by rendezvous with an orbital spacecraft in lunar orbit.

The study is directed toward the development of conceptual arrangements for a variety of devices including man supported and self supporting configurations with varying ΔV capability, and toward definition and documentation of their physical and dynamic properties in a manner which permits their use as inputs to the simulation studies planned to be conducted by NASA for final evaluation and selection of one or more of these configurations.

Since the results of the simulation studies cannot be foretold, it was deemed necessary to present parametric data and analytical methods to allow iterative steps to be made which will permit changes to the furnished configurations as dictated by the results of simulation while maintaining the ground rules and design criteria which were employed in the initial system definition.

The study is further directed toward the development of "minimum complexity" devices which make extended utilization of the man in the system. The purpose of this effort is to provide inputs for evaluation by NASA through simulation studies of the ability of the man to control these vehicles in their minimum complexity form, establish the task loading of the operator on a variety of transportation and escape missions, establish the guidance and navigation equipment for successful completion of the mission and conduct tradeoffs between energy management and sensor requirements.

B. SUMMARY

This report describes the system requirements for typical missions, outlines the basic assumptions and ground rules, and establishes nominal vehicle designs to achieve these requirements. The results of the analyses provide information for tradeoffs in the areas of stabilization and control parameters, propulsion systems, structures and equipment; identify problem areas and delineate the remedial actions taken for their solution during the course of the study. The tradeoffs in the study have been conducted without the aid of simulation studies. Further refinements are possible only with inclusion of such studies together with definition of specific mission profiles to which the systems will be subjected.

The body of the report is organized into seven sections which deal with control and stabilization, propulsion, tanks and bladders, electrical and electronic, design criteria configuration studies, and simulation recommendations.

Section II contains the stabilization and control system analyses and constraints and develops equations for evaluating three basic modes of control; main engine gimballing, differential throttling, reaction control jets or combinations of these.

The propulsion section presents performance, envelope and weight plots for radiation-cooled and ablative thrust chambers using N_2O_4/0.50 N_2H_4 + 0.50 UDMH propellants over a thrust range of 100 to 2500 pounds for fixed thrust and throttleable chambers capable of throttling across a 10:1 range. All propulsion system analyses and chamber designs considered have multiple restart requirements. Radiation-cooled chambers have a cumulative burn time exceeding one hour whereas ablative chamber design characteristics were defined as a function of their cumulative burn time. The propellants investigated for application to reaction control systems included the cold gases, monopropellants and liquid and gaseous bipropellants. Performance curves for steady-state and pulsed operation, system schematics and component weights are presented in parametric form for each of the candidate propellants. Volume II (Confidential) of this report presents parametric design and performance data for high-energy solid propellants which are applicable to the boost phase of the escape devices.

A summary of current and advanced positive expulsion techniques is presented in Section IV of the report, together with considerations for tank geometry and applicable materials for use on the Personnel Propulsion Devices. Parametric plots of tank and bladder weights are presented as a function of working pressure, tank volume and length to diameter ratio for tanks fabricated of the selected material.

A survey was made of the available guidance, navigation and control, equipment and sensor applicable to the Personnel Propulsion Devices. The equipment is assembled into four groups arranged in ascending order of sophistication and the weight and power requirements for each group is established. In order to permit equipment substitution, should such action become necessary during the simulation studies, a listing of equipment is also presented together with their physical and functional characteristics, power requirements and their dynamic range of coverage.

The loads incurred during transportation, test and operation of the Personnel Propulsion Devices were identified and design criteria established. Based on these criteria, the airframe weights were estimated and materials were selected for vehicle construction. A simple spring type landing gear constructed of tubular fiber glass has been investigated because it offers three distinct advantages over other types of gears: (1) it offers an inherent design and operating simplicity (2) it is completely reusable and (3) it has no moving parts. Although the spring gear does not meet the maximum landing velocity criteria, it has enough merits to justify further investigation through simulation studies.

One- and two-man devices have been configured to provide short-, intermediate- and long-range transportation for lunar-based personnel and to provide escape capability from the surface of the moon into an 80-100 nautical mile lunar orbit. The results of these configuration studies are found in Section VII. The configurations developed fall within one of the following general categories:

(1) Back Packs - Devices in which the propulsion system is carried by the man (applicable to one-man short- and intermediate-range transportation).

(2) Platforms - Devices which accommodate the man in standing position and permit attitude control by kinesthetic means (applicable to one-man, intermediate- and long-range transportation).

(3) Vehicles - Devices which provide a seating arrangement for the man (applicable to long-range transportation and in escape missions).

Consistent with the ground rules set forth for this study, all configurations developed for the Personnel Propulsion Devices are "minimum configurations"; that is, they incorporate a minimum of automatic equipment. The pilot aided by simple sensors is the sole monitor and controller of the vehicle. However, since simulation studies were beyond the scope of this study, it is not known how close to overloaded the pilot may be during operation of the vehicles. The requirements and constraints developed in the earlier sections of this report were used to screen the conceptual configurations. The configurations which survived the initial screening were further refined. A weight and inertia table is provided for each candidate configuration for use in the simulation studies.

The final section of this report presents recommendations for the conduct of the simulation studies. Simulation schematics are presented for vehicle control by gimballing the main engines, differential throttling of the main engines or by using a set of reaction control jets. In the latter, provisions have been made for evaluation of the various command modes, i.e., acceleration, rate and rate with position hold, by synthesis of individual schematics.

C. CONCLUSIONS

Due to the scope of this effort, which excluded the use of simulation and trajectory studies, the conclusions presented below are based on the physical characteristics of the devices presented and therefore must be considered "conditional" until such time as they can be verified by means of simulation studies applied to specific mission profiles.

(1) Large center-of-gravity excursions were found to be prohibitive on simple devices without the use of self compensating networks to either gimbal translate or proportionally throttle the main engine(s).

(2) Approximately 10:1 effective throttling is required for trajectory control of transportation devices having an initial earth thrust-to-weight ratio of 0.5. No throttling capability is required for the escape mission.

(3) With large center-of-gravity excursions compensated through judicious placement of consumables, rigid main engines with reaction control jets have been found to offer the most efficient, least weight/complexity system for use on the Personnel Propulsion Devices.

(4) Prelaunch alignment of the thrust vector with the center of gravity is necessary to prevent overturning of the vehicle at launch and minimize propellant consumption for attitude control during flight.

(5) The variation in launch weight of any transportation device operating at the optimized chamber pressure (80 psia), due to change in main engine area ratio between 40 and 100, results in less than 1% penalty in launch weight at the lower area ratios.

(6) Radiation-cooled thrust chambers provide widest mission flexibility with respect to burning time, eliminate outgasing in vacuum during "off" periods in the heated condition, eliminate the throat erosion, and thrust misalignment problems present in ablative chambers and result in minimum vehicle weight.

(7) Solid propellants of high specific impulse are desirable for application to first-stage boost of escape devices in view of their inherent simplicity. A liquid second stage however, is required to supplement the characteristic velocity and to provide restart capability for orbit injection and terminal maneuvers. Staging from solid to liquid disclosed no significant weight advantage at any point between 3000 and 6000 fps ΔV for escape vehicles with total ΔV capability of 8000 fps. Thrust level and burning time impose limitations in the application of solid propellants to vehicles of low thrust-to-weight ratio and large total impulse.

(8) The use of N_2O_4/0.50 N_2H_4 + 0.50 UDMH propellant combination in the attitude control system of the vehicle results in the least weight penalty to the vehicle based on a total impulse requirement of 8000 lb-sec delivered at 100 millisecond pulsewidth. The N_2O_4 system also provides the widest system flexibility and least I_{sp} degradation when operated in pulse mode for equivalent square wave impulse $>$ 10 milliseconds.

(9) Considerable weight savings can be realized in the transportation devices from the use of cantilevered spring landing gears, if the touchdown velocities can be reduced within acceptable limits (3, 1.5, 1.5 fps) through intensive pilot training and/or the addition of stability augmentation during the landing maneuver.

(10) Back pack configurations for transportation provide considerable weight savings over vehicle or platform configurations within the ΔV range of 2000-4000 fps.

(11) Use of two back packs to provide the same ΔV capability for two men results in approximately equal total weight and smaller total storage volume than a two-man vehicle. However, the back pack configurations lack the capability to transport a passenger astronaut or equivalent payload.

(12) The gross weight difference between a dual function device capable of multiple transportation and escape missions and a single mission escape device at the same thrust-to-weight ratio is approximately 100 pounds in favor of the escape device.

(13) Mission flexibility is attainable in the transportation or dual function devices by employing off-loaded vehicles designed for larger ΔV increments in short excursions. The study results indicate that less than 100 pounds difference exists in the burnout weight between a device designed for 2000 fps ΔV and one designed for 6000 fps ΔV in favor of the vehicle with the smaller ΔV capability.

(14) Simulation studies are required to evaluate the effects of the following parameters.

(a) Pilot task loading during flight to include the tasks of vehicle control guidance and navigation.

(b) Mode of control--i.e., acceleration command, rate command or rate command with position hold feature, as it relates to:

(1) The mission

(2) Vehicle controllability

(3) Energy management

(c) The effect of unbalanced moment producing devices on vehicle controllability, cutoff ΔV and cross coupling in a six degree-of-freedom simulation.

(d) The navigation and guidance requirements for transportation and escape missions and the equipment and displays sequenced for successful completion of the intended missions.

(e) The optimum torque-to-inertia ratio for manually controlled vehicles with specific emphasis placed on the landing phase of transportation vehicles and the docking phase of escape vehicles.

(f) The magnitude and repeatability of residual velocities at landing.

(g) Manual translation control by means of orientation of a single longitudinal thrust axis, as opposed to orthagonally applied thrust (vertical and horizontal).

SECTION II

SYSTEMS ANALYSIS

A. GENERAL

This section contains parametric attitude stabilization and control data generated in support of the general PPD configuration study. In particular, this section presents the tradeoff relationships and constrains necessary for a logical screening operation on the various possible vehicle configuration concepts and the systematic selection of workable vehicle configurations.

Data indicating the effects of variations in the attitude control systems of the selected vehicles are also presented to facilitate the simulation of various modifications of the selected configurations.

The purpose and applications of the results of this study are identified in the various presentations throughout this report.

Three basic methods for producing attitude control moments are discussed.

(1) Main engine gimballing or translating.

(2) Differential throttling in a multiple engine configuration.

(3) Set of fixed thrust reaction jets.

In the following development the three different moment producing methods were treated separately - i.e. it was assumed that a vehicle has only one of the three means for attitude control and stabilization. The various relationships and constrains obtained in this manner are, however, applicable as well to mixed configurations employing two or all of the three moment producing methods in any one axis.

The relationships and constrains obtained are independent of the closed loop control system used for any particular vehicle configuration. A separate section of this study deals with the closed loop attitude control of varying degrees of complexity.

B. GIMBALLED MAIN PROPULSION UNIT

1. General

The material presented here is based on a single engine vehicle configuration gimballable in one axis. All relationships, however, are applicable with slight modifications to vehicle configurations with gimballed engine clusters, multiple engine configurations gimballed in a coordinated manner, and mixed configurations where rigid mounted as well as gimballed engines are incorporated.

Figure 1 defines the symbols used in this presentation.

The main advantage of using the gimballed main engine for attitude control is the saving in hardware and propellant requirements associated with using a separate set of attitude control reaction jets. Another advantage is that proportional (continuously variable) angular accelerations are provided, rather than the on-off accelerations associated with reaction jets.

The disadvantages of using the gimballed main engine for attitude control are the need for a gimbal actuating mechanism and the corresponding power requirements, the vehicle structure requirements to allow clearance for the gimballing action, the varying orientation between the thrust vector and the vehicle body axes, and the lack of attitude stabilization during in-flight main-engine-off periods. For a single engine configuration, lift engine gimballing will provide pitch and roll control - for yaw control a set of two or four reaction jets must be provided.

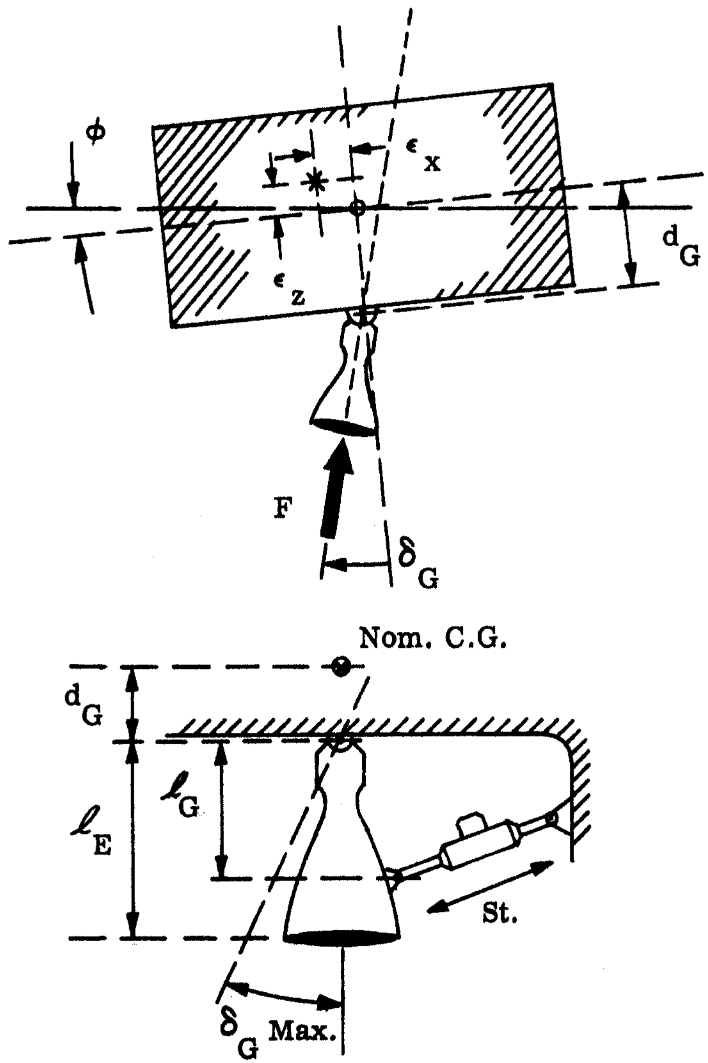
2. Vehicle Stabilization

Various disturbance moments will act on the vehicle during its flight; these disturbances may be caused by a vehicle unbalance (gradual or sudden c.g. shifts away from the nominal c.g. location), by erratic gimbal action, or both. This development considers only disturbances normally to be expected (shifting of the actual c.g. due to propellant burn-off and unequal tank drainage). Disturbances caused by various possible malfunctions in the vehicle, the gimbal mechanism, and (or) the control loop are not directly treated here, although their effect can be inferred from the data presented.

It is assumed that the vehicle at launch is properly balanced (Section II.E.4 deals with this respect of the mission). When the vehicle is in powered flight there may occur a show shifting of the c.g. due to propellant burn-off, and the resulting error moment and its effects will be corrected by gimballing the main propulsion unit. The gimbal angle required for c.g. shift compensation and the gimbal angle resolution needed to properly accomplish this depends on the vehicle dimensions, maximum thrust level, moments of inertia, maximum c.g. travel, and the permissible level of attitude acceleration uncertainty.

a. Gimbal Range Requirements for c.g. Shift Compensation

Any vehicle configuration that uses gimballing of the main propulsion unit for c.g. shift compensation must satisfy the relationship between the required gimbal angle ($\delta_{G\ (c.g.)}$), the gimbal arm (d_G) and the maximum c.g. shift (ϵ_{max}) as shown in Figure 2. A vehicle configuration providing a particular d_G and permitting some ϵ_{max} must also provide at least the indicated $\delta_{G\ (c.g.)}$ for moment balance. Thus Figure 2 may be used to screen out those vehicle configurations which do not lend themselves to vehicle stabilization by main engine gimballing.



F	Main Engine Thrust	lb
Δ F	Thrust Uncertainty	lb
d_G	Distance Gimbal Point - c.g.	ft
ϵ_x	Horizontal c.g. Shift	ft
ϵ_z	Vertical c.g. Shift	ft
δ_G	Gimbal Angle	deg
$\Delta\delta$	Gimbal Angle Uncertainty	deg
ℓ_E	Length of Main Thruster	ft
ℓ_G	Gimbal Actuator Moment Arm	ft
St	Actuator Stroke	ft
θ	Vehicle Attitude	deg
I	Moment of Inertia	slug-ft^2

Figure 1. Gimballed Main Propulsion Unit

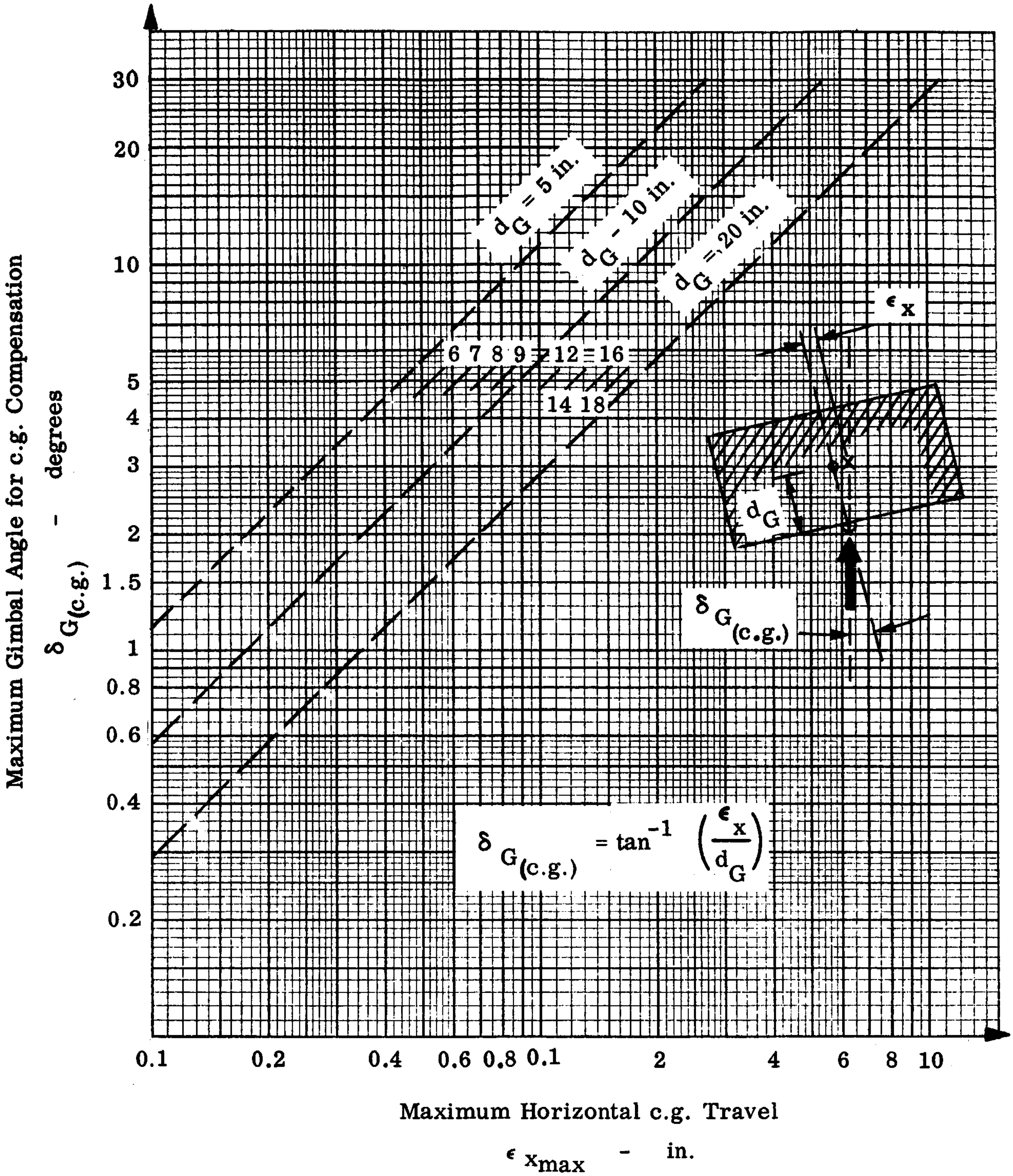


Figure 2. Gimbal Range Requirements for c.g. Shift Compensation

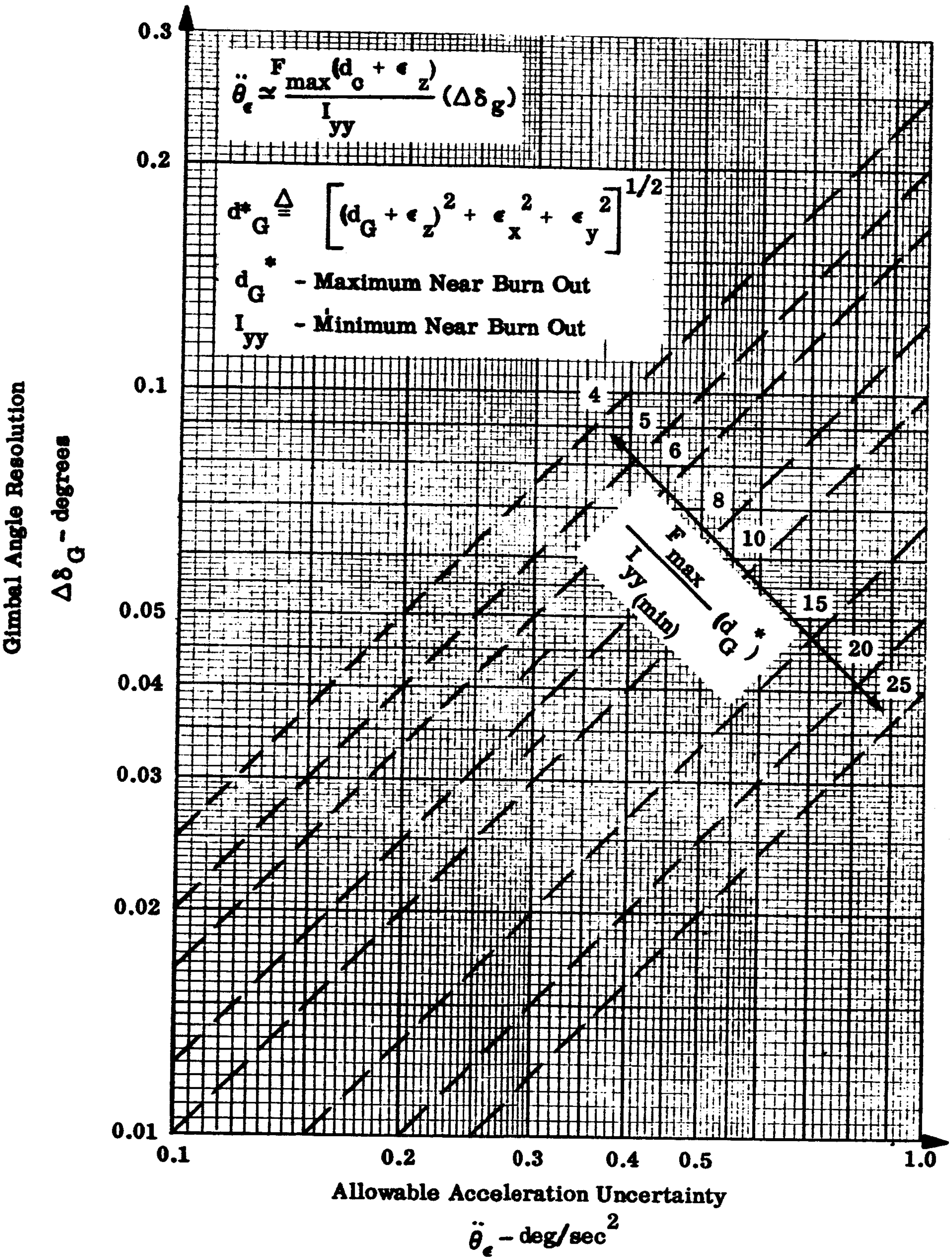


Figure 3. Gimbal Resolution Requirements in Terms of Attitude Acceleration Uncertainty

b. Gimbal Angle Resolution Requirements

For a given vehicle configuration (known values of F_{max}, I_{min}, d_G, and ϵ) and some specified maximum attitude acceleration uncertainty ($\ddot{\theta}$ error) based on guidance and control considerations the permissible gimbal angle uncertainty or gimbal angle resolution ($\Delta\delta_G$) is determined.

$$\ddot{\theta}_{Error} \simeq \left(\frac{F}{I_{yy}}\right) d_G^* \ \Delta\delta_G$$

$$d_G^* \triangleq \sqrt{(d_G + \epsilon_z)^2 + \epsilon_x^2 + \epsilon_y^2}$$

Figure 3 shows this relationship and can be used to determine the required gimbal angle resolution for a given vehicle lay-out. Configurations which require extremely precise control may be screened out from further consideration on this basis.

Any attitude acceleration uncertainty will in general cause a limit cycle about some nominal attitude position. The magnitude and frequency of this oscillation, and the resultant degradation in the vehicle performance are largely determined by the details of the attitude loop-closing mechanism.

c. Thrust Misalignment and Cross-Coupling

If c.g. shifts during main engine firing are compensated by means of main engine gimballing the nominal vehicle attitude will change; the thrust magnitude and direction is determined only by the flight program or flight requirements, and thus c.g. shift compensation consists essentially of appropriately rotating the vehicle about the thrust vector as illustrated in Figure 4.

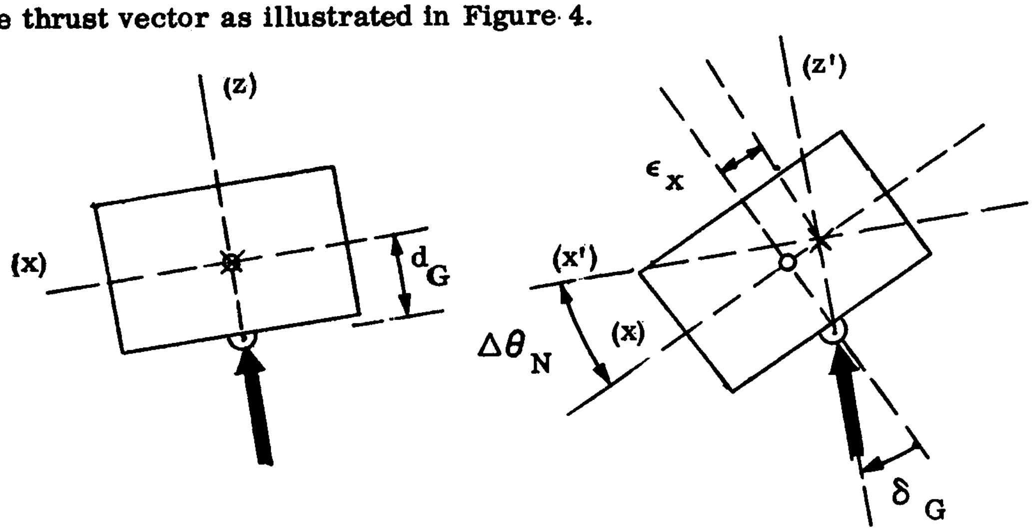


Figure 4. Nominal Attitude Changes with c.g. Shift

The tilting of the vehicle must be considered in the various control and guidance schemes that make use of equipment fastened to the vehicle body (sensors, optical sights, etc.).

Due to the change in nominal vehicle attitude ($\Delta\theta_n \simeq \epsilon_x/d_G$) there will occur Pitch-Roll-Yaw coupling for commanded attitude changes.

The coupling effects may be expressed in form of a transformation matrix which relates the orthagonal x-y-z system to the shifted x'-y'-z' system.

$$\begin{Bmatrix} x' \\ y' \\ z' \end{Bmatrix} = \begin{pmatrix} \cos\alpha & 0 & \sin(\alpha) \\ -\sin(\alpha)\sin(\beta) & \cos(\beta) & \cos(\alpha)\sin(\beta) \\ -\sin(\alpha)\cos(\beta) & -\sin(\beta) & \cos(\alpha)\cos(\beta) \end{pmatrix} \begin{Bmatrix} x \\ y \\ z \end{Bmatrix}$$

where the terms are defined as follows:

$$\sin(\alpha) = \frac{\epsilon_x}{\sqrt{(d_G + \epsilon_z)^2 + (\epsilon_x)^2}}$$

$$\cos(\alpha) = \frac{(d_G + \epsilon_z)}{\sqrt{(d_G + \epsilon_z)^2 + (\epsilon_x)^2}}$$

$$\sin(\beta) = \frac{\epsilon_Y}{\sqrt{(d_G + \epsilon_z)^2 + \epsilon_x^2 + \epsilon_y^2}}$$

$$\cos(\beta) = \frac{\sqrt{(d_G + \epsilon_z)^2 + (\epsilon_x)^2}}{\sqrt{(d_G + \epsilon_z)^2 + \epsilon_x^2 + \epsilon_y^2}}$$

If moments are represented in vectorial form – $\ddot{\theta}$ vector along y-axis, $\ddot{\phi}$ vector along x-axis, $\ddot{\psi}$ vector along z-axis – then the above matrix defines the control coupling relationships.

$$\ddot{\phi}_{coupled}/\ddot{\psi}_{command} = \sin(\alpha), \quad \ddot{\psi}_{coupled}/\ddot{\theta}_{command} = -\sin(\beta),$$

etc.

3. Commanded Maneuvering

a. Control Power

The attitude acceleration (control power) available for a vehicle configuration which uses its gimballed main propulsion unit for attitude control in pitch and roll depends on the vehicle dimensions, the thrust level, the moments of inertia, and the gimbal angle range. For the pitch acceleration the following equation illustrates the relationship.

$$\ddot{\theta}_{command} = \frac{F}{I_{yy}} d_G^* \sin(\delta_{G_{command}})$$

The available control power at any time during the mission will depend on the operating thrust level and the values of d_G^* and I_{yy} at the particular time. If lift-off is achieved with maximum thrust, and if the landing is made at the minimum thrust level (near burnout condition) then the following expressions for $\ddot{\theta}_{command}$ can be written.

$$\text{Launch} \qquad \ddot{\theta}_{command} = \frac{F_{max}}{I_{yy(max)}} \left(d_G\right) \sin \delta_{G_{command}}$$

$$\text{Landing} \qquad \ddot{\theta}_{command} = \frac{F_{min}}{I_{yy(min)}} \left(d_G^*\right) \sin \delta_{G_{command}}$$

Different thrust programs will result in different extremes.

If the attitude control power for a given vehicle configuration is required to be equal to or greater than some $\ddot{\theta}_{c(min)}$ then the above expressions may be used in determining the required gimbal range. Figure 5 presents the general relationship between the control power, the gimbal range, and the vehicle parameters, and can be used in selecting the required gimbal angle range.

For a given vehicle configuration (known F, I, d_G, and ϵ) and for a specified control power requirement, Figures 3 and 5 can be used to determine the total required gimbal range for attitude stabilization and control.

$$\delta_{G(total)} = \delta_{G(c.g.)} + \delta_{G(command)}$$

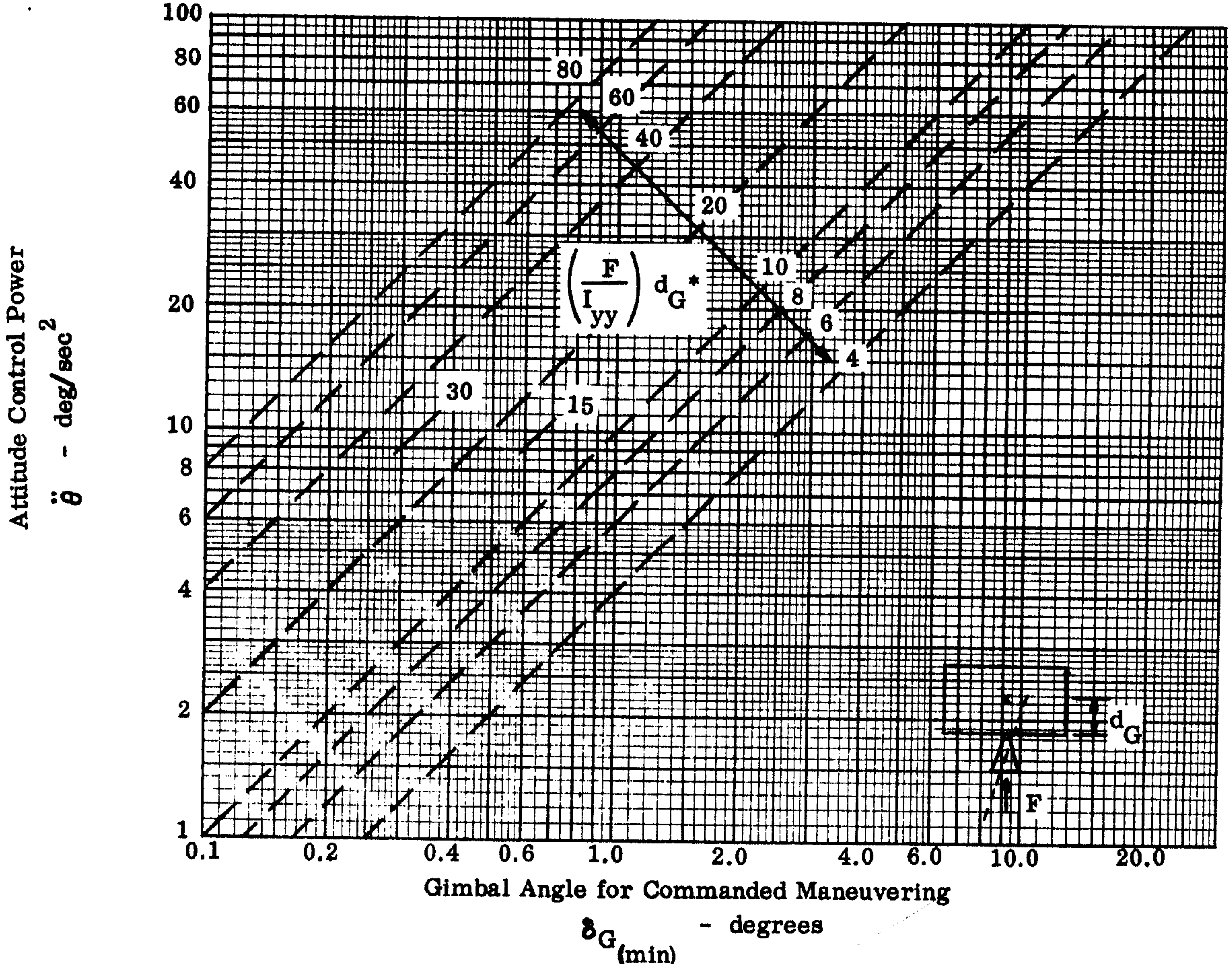
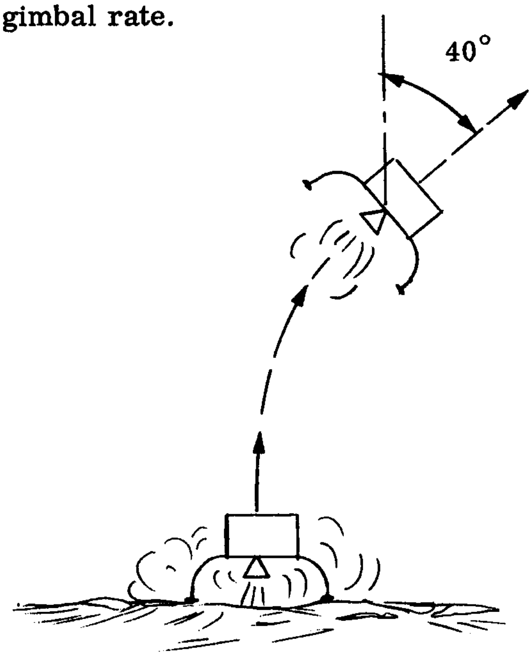


Figure 5. Gimbal Range Requirements For Commanded Attitude Acceleration

b. Required Gimbal Acceleration, Gimbal Velocity, and Gimbal Range to Execute a Simple Attitude Maneuver in a Given Time

The speed at which a desired attitude maneuver can be executed will depend largely on the dynamic response of the gimbal mechanism. For a small maneuver to be executed in a brief time the gimbal acceleration will be the limiting factor; for larger maneuvers the maximum obtainable gimbal rate (saturation condition) will be the determining factor.

A typical mission will require a number of commanded attitude maneuvers for the purpose of course alteration, corrections, transition from a vertical launch trajectory to some inclined boost trajectory, etc. Considerations of mission efficiency may require that a certain turning maneuver must be executed within a given time; a specification of this sort would then require a certain gimbal acceleration or gimbal rate.



Example of a Vertical Lift-Off Command Turn (0° - 40°) to Achieve Typical Boost Trajectory for a Ballistic Flight

Figure 6. Command Turn During Launch Phase

Figures 7 and 8 indicate the accelerations and rate requirements respectively to perform the typical 40° turning maneuver illustrated in Figure 6, as a function of the time in which the maneuver is to be performed. The curves on Figures 7 and 8 can easily be adjusted for commanded maneuvers other than 40° by merely shifting the whole family of curves vertically by the proper amount.- for $\theta_{command} < 40°$ shift down by $(\theta_{command}/40)$.

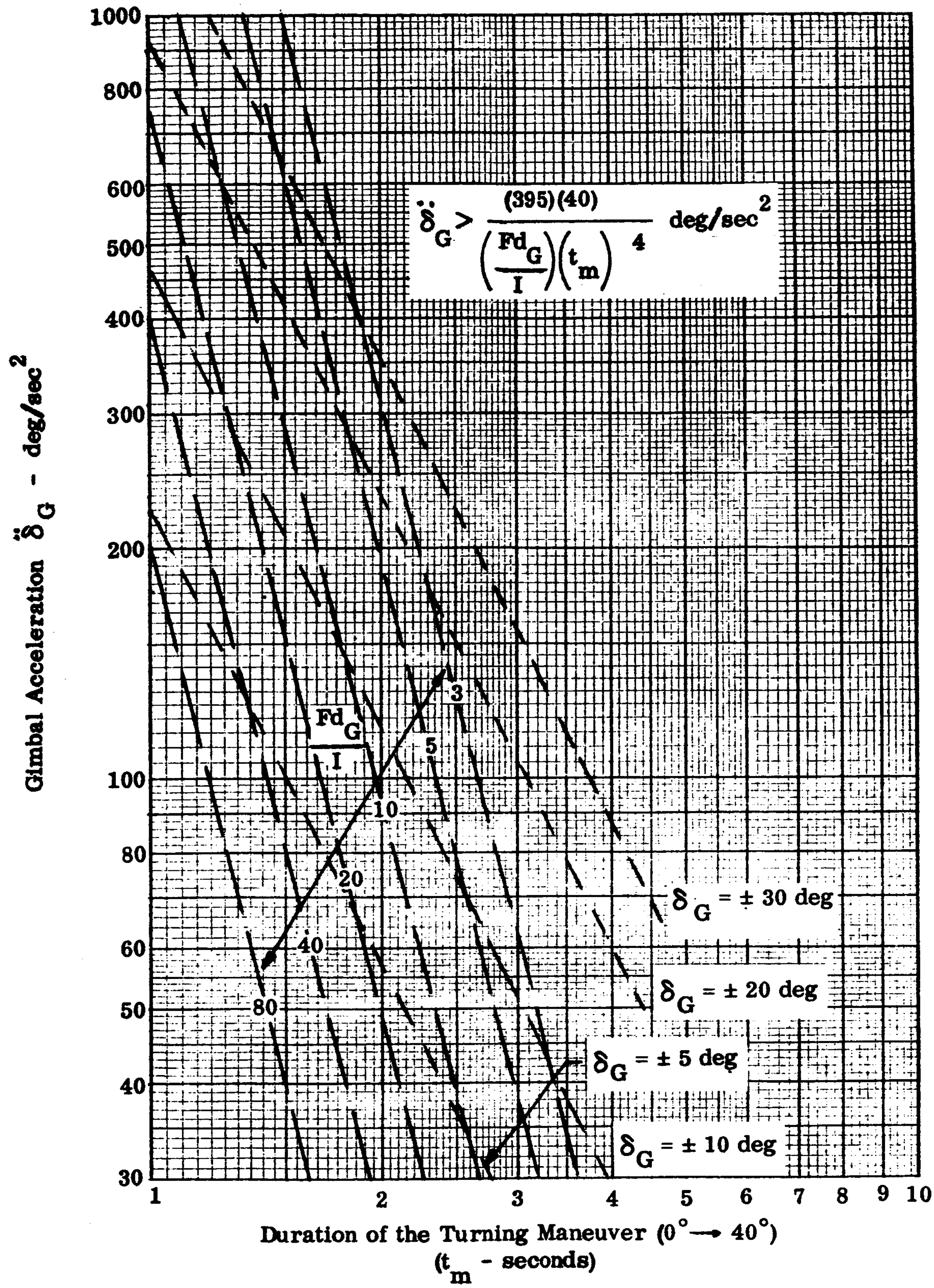


Figure 7. Gimbal Acceleration Requirements for Minimum Time Attitude Turn of 40°

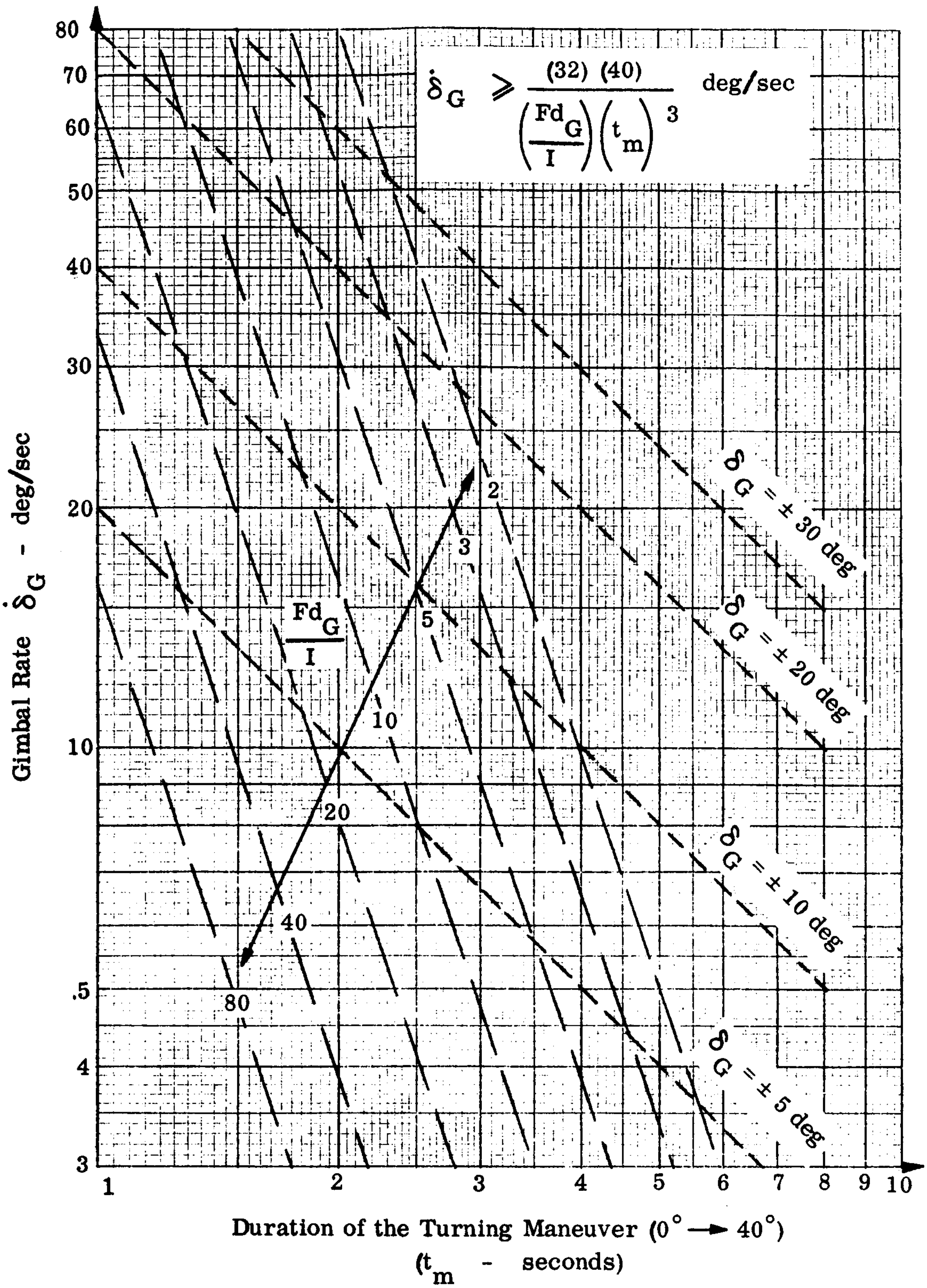


Figure 8. Gimbal Velocity Requirements for Minimum Time Attitude Turn of 40°

4. Yaw Maneuvering by Means of Differential Gimballing in a Two-Engine Configuration

A gimballed single-engine configuration cannot achieve yaw by means of main engine gimballing, but a vehicle with two gimballed engines may obtain yaw moments by differential gimballing, as well as pitch and roll by collective gimballing. Figure 9 illustrates schematically a two-gimballed engine arrangement.

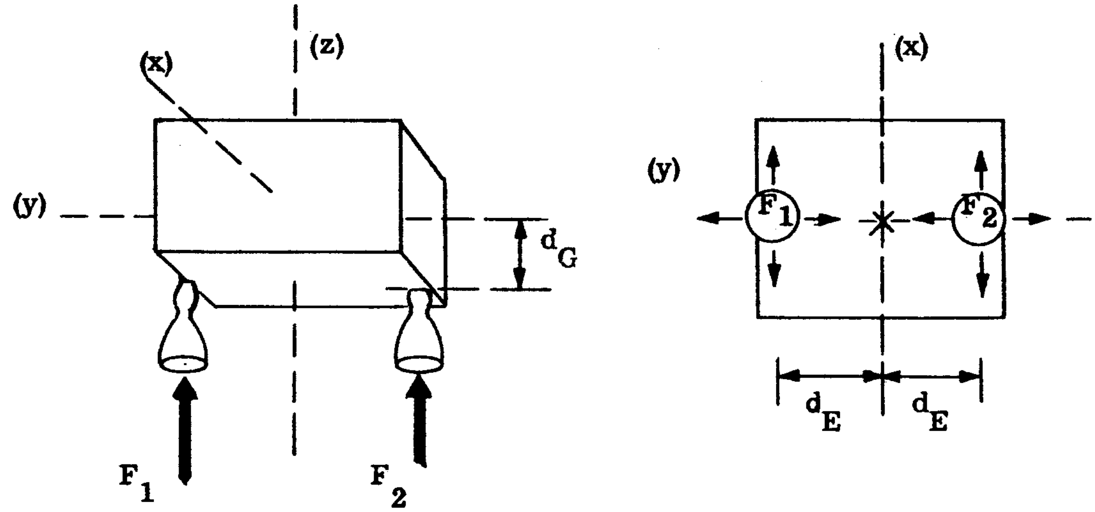


Figure 9. Gimballed Two-Engine Configuration

The relationship between the yaw-control power and the vehicle parameters and dimensions is given in Figure 10; this relationship may be applied in determining the required gimbal angle range corresponding to some specified control power.

To determine the total gimbal range for this two-engine configuration, one or more of Figure 2, 5, 7, 8, and 10 may be used.

$$\delta_{G_{(total)}} = \delta_{G_{(stabilize)}} + \delta_{G_{(pitch)}} + \delta_{G_{(yaw)}}$$

C. DIFFERENTIAL THROTTLING FOR RIGIDLY MOUNTED 2- OR 4-ENGINE CONFIGURATIONS

1. General

The material presented here is based on a single-axis analysis for a simple 2-engine vehicle configuration. All relationships developed here are also applicable to configurations involving other multiple engine arrangements, or to mixed configurations with gimballed as well as throttleable engines.

Figures 11 and 12 define the symbols used in this presentation.

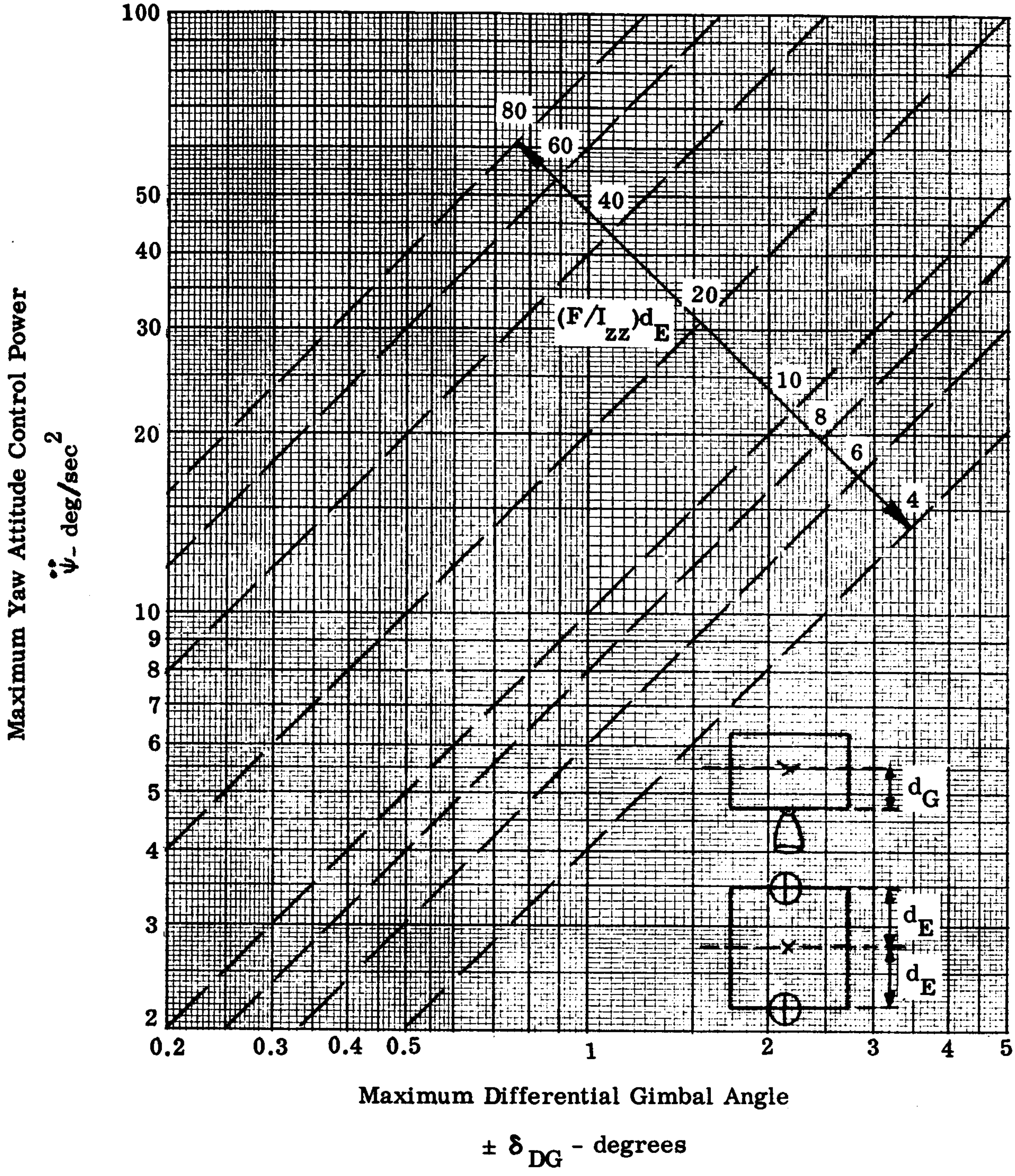
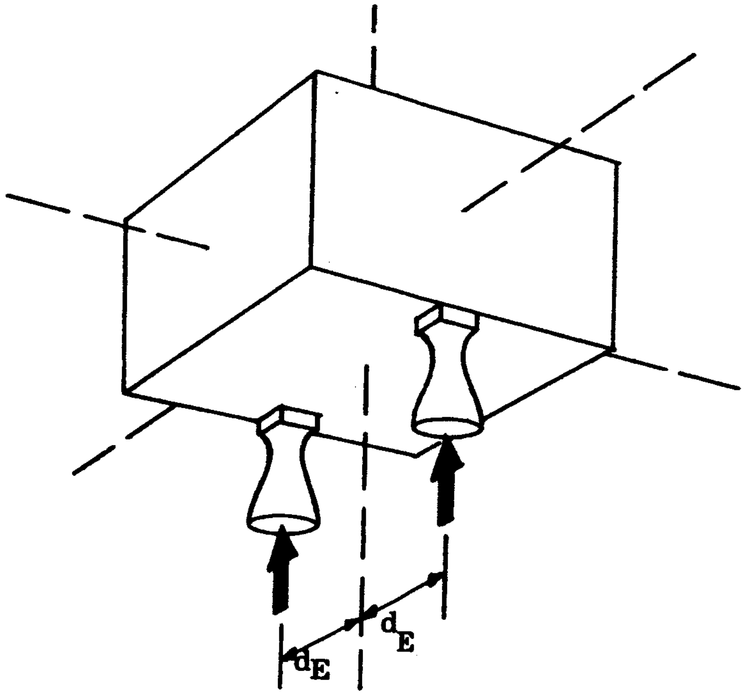


Figure 10. Gimbal Range Requirement for Yaw Control Power



F_{max}	Total Maximum Thrust	lb
F_{min}	Total Minimum Thrust	lb
F_{op}	Operating Thrust ($F_{min} < F_{op} < F_{max}$)	lb
ΔF	Individual Engine Thrust Uncertainty	lb
T.R.	Throttling Ratio	-
d_E	Moment Arm (2 Engine or Square 4 Engine)	ft
d_T	Moment Arm (Diamond 4E)	ft
ϵ	c.g. Shift	ft
I_{ii}	Moment of Inertia (ii - Axis)	slug-ft^2
I_{sp}	Specific Impulse	$\frac{\text{lb-sec}}{\text{lb}}$

Figure 11. Simple 2-Engine Configuration

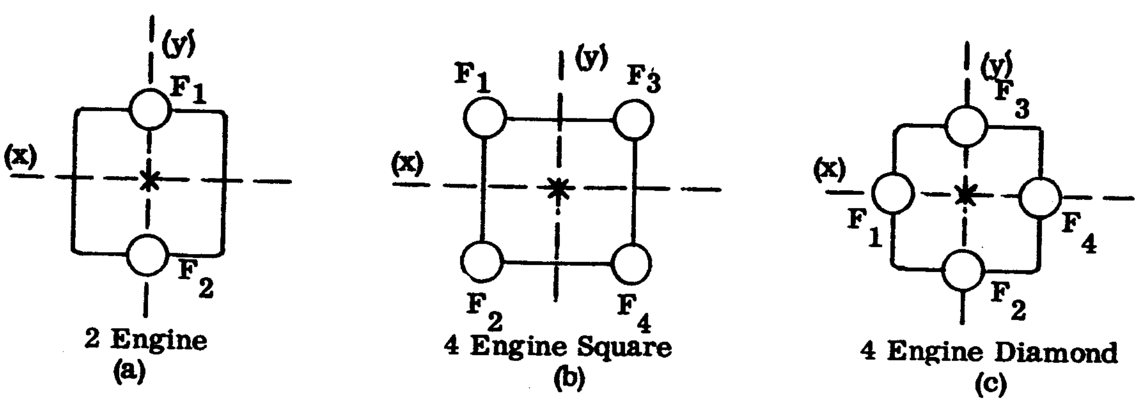


Figure 12. Multiple Main Thruster Arrangements

For a multiple engine configuration with collective throttle capability, the addition of differential throttle capability for pitch and roll attitude control and stabilization demands relatively little additional hardware. Unlike the gimballed configuration, the throttle stabilized vehicle attitude does not vary with horizontal c.g. shifts, and vertical c.g. shifts do not affect the control system.

The limitations and shortcomings of this method are the following. At least two throttleable engines, separated by a distance $2d_E$ are required to obtain control in a single axis - thus the 2-engine configuration in Figure 12a would only provide for roll control, the 4-engine configurations of Figure 12b and c provide pitch and roll. Yaw control must be achieved by means other than differential throttling of the man engines. Differential throttling will increase the required throttling range of the engines and will add to the I_{sp} degradation caused by the collective throttling. The thrust uncertainties ($\pm\Delta F$) will induce error moments ($M_\epsilon \leqslant 2d_E\Delta F$) and thus will set a lower bound on the accuracy obtainable with this method. During main engine off-periods, as would occur during the coast phase in a ballistic flight, attitude stabilization cannot be provided by this method.

2. Vehicle Stabilization

Disturbance moments caused by c.g. shifts or by differential thrust uncertainties may act on the vehicle during its powered flight. In this presentation, only the effects of normal c.g. shifts due to propellant burn-off and the effects of the small thrust uncertainties are considered. Disturbances caused by various possible malfunctions in the vehicle (sudden drastic c.g. changes, failure of one or more thrusters, etc.) are not specifically treated here, but the extension to these cases is straightforward. It is assumed here that the vehicle at launch is properly balanced - Section II.D.4 deals with this aspect of the mission.

The collective throttle setting will be dictated by the flight program or mission requirements such that the total operating thrust is at some value between the maximum and minimum thrust possible.

$$F_{min} < F_{op} < F_{max}$$

When the vehicle is in powered flight only the horizontal c.g. shifts - c.g. shifts along the body x-axis and y-axis (ϵ_x, ϵ_y) - require differential throttling action. Differential throttling for attitude control and stabilization will not alter the value of F_{op} - exceptions to this rule will occur only for flights requiring an operating thrust equal to either the maximum thrust on the minimum thrust.

a. C.G. Shift Compensation

For a 2-engine configuration, operating at some $F_{op} = F_1 + F_2$, the differential throttling ratio (F_1/F_2) can be expressed in terms of the c.g. shift (ϵ) and the engine separation ($2d_E$).

$$\frac{F_1}{F_2} = \frac{\frac{F_{op}}{2} - F_{D.T.}}{\frac{F_{op}}{2} + F_{D.T.}} = \frac{1 - \frac{\epsilon}{d_E}}{1 + \frac{\epsilon}{d_E}}$$

If the operating thrust (F_{op}) is equal to F_{max} then any differential throttling requirement will result in a decrease of F_{op}; similarly if the operating thrust requirement calls for F_{min} any differential throttling will result in an increase in F_{op}.

For $\epsilon \neq 0$

Maximum Collect. Thrust: $$F_{op} = \frac{F_{max}}{2} + \left(\frac{F_{max}}{2} - F_{D.T.}\right) < F_{max}$$

Minimum Collect. Thrust: $$F_{op} = \left(\frac{F_{min}}{2} + F_{D.T.}\right) + \left(\frac{F_{min}}{2}\right) > F_{min}$$

Figure 13 shows the differential throttling ratio (F_1/F_2) and the thrust loss in % of F_{max}, which would occur if maximum collective thrust is called for, plotted versus the ratio of c.g. shift to engine separation (ϵ/d_E).

On the basis of practical considerations and general mission requirements an individual engine throttling ratio of 10:1 was specified. With this restriction, the maximum differential throttling ratio is also fixed at 10:1

$$\frac{F_1}{F_2} \geqslant \frac{\frac{F_{min}}{2}}{\frac{F_{max}}{2}} = \frac{1}{10}$$

This in turn determines the absolute maximum value which the ratio ϵ/d may assume.

$$\left(\frac{\epsilon}{d_E}\right)_{max} = \frac{9}{11}$$

The only operating thrust possible under the above extreme conditions will be

$$F_{op} = \frac{F_{max}}{2} + \frac{F_{max}}{(2)(10)} = 0.55\ F_{max}$$

That is, the available collective throttleability is reduced to 1:1 by complete differential throttling of 10:1. Figure 14 shows corresponding curves for a 4-engine configuration. Any horizontal c.g. shift will be some combination of Case I and Case II as illustrated in Figure 14, and any corresponding differential throttling curve will lie in the region bounded by Curves I and II.

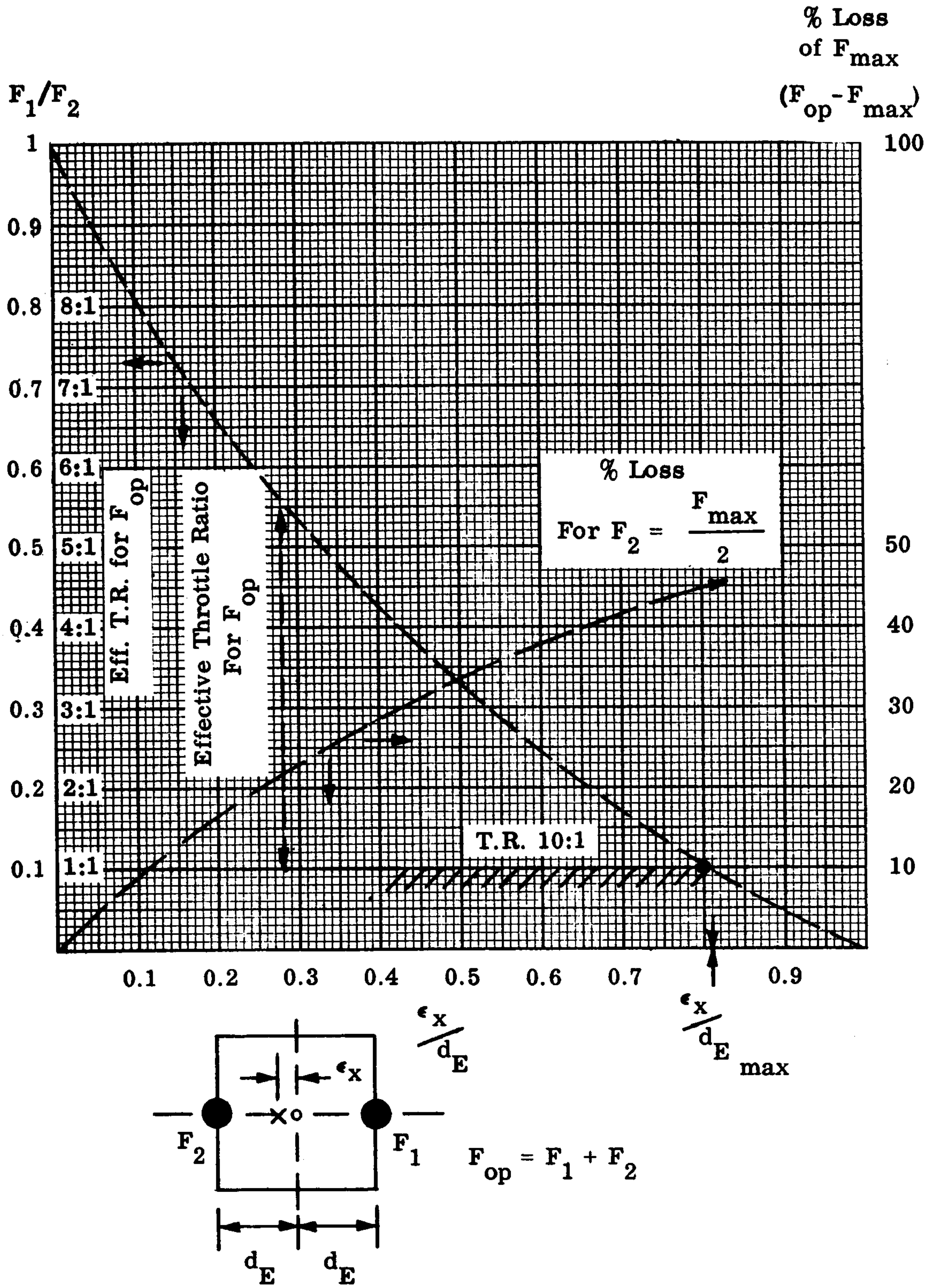


Figure 13. Differential Throttling to Correct c.g. Shift Effects (2 Engine Configuration)

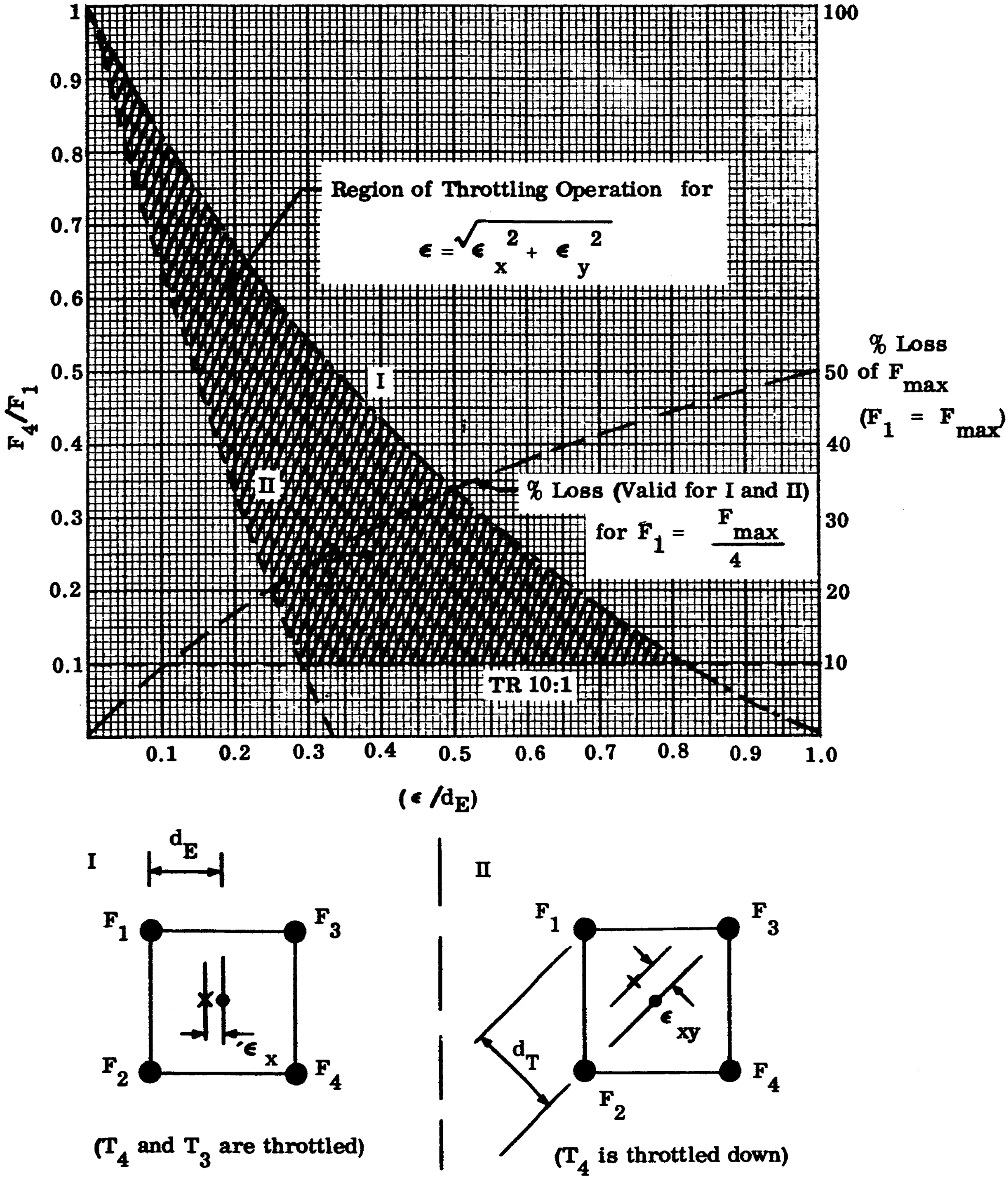


Figure 14. Differential Throttling to Compensate c.g. Shifts

Any differential throttling will reduce the available collective throttle-ability. If a specified nominal collective throttling ratio ($T.R._{collect.}$) is to be maintained in addition to maintaining capability for differential throttling, and increased individual engine throttling ratio ($T.R._{indiv.}$) must be provided.

Figure 15 shows the relationship between $T.R._{collect.}$ and $T.R._{indiv.}$ as a function of the engine separation and the maximum expected c.g. shift.

b. Thrust Uncertainty

For all stabilized conditions there will be in general a small residue error moment due to the thrust uncertainty ($\pm \Delta F$).

Figure 16 shows the resulting error acceleration as a function of the vehicle parameters and the % thrust uncertainty. The desired acceleration resolution (or maximum allowable error acceleration) and a given obtainable thrust uncertainty will indicate the required value of the engine separation, vehicle inertia, and the maximum thrust level.

3. Commanded Maneuvering

The attitude control power available for a vehicle with zero c.g. shift depends on the collective operating thrust level, the collective throttling ratio, and the physical parameters of the vehicle. Since any commanded attitude acceleration is obtained by commanded differential throttling such that the operating collective thrust level is not altered, it is apparent that no control power is available for operating collective thrust levels equal to F_{min} or F_{max}, and that absolute maximum control power is available when $F_{op} = \frac{F_{max}}{2} + \frac{F_{min}}{2} = 0.55\, F_{max}$ for $T.R._{collective} = 10:1$. Figure 17 shows the maximum available control power in terms of the collective operating thrust level (in % of F_{max}) and the physical parameters of a given vehicle. From Figure 17 it is seen that control power of the order of 10 deg./sec^2 or more may generally be obtained if F_{op} is kept between about 90% and 20% of F_{max}.

From the above considerations it appears that for the vehicle configurations examined here it will be the acceleration resolution, and not the maximum acceleration capability which will be the limiting performance factor.

For the presence of a sustained disturbance moment, such as caused by a c.g. shift, the maximum available control power will be reduced in one direction and increased correspondingly in the other direction. The condition is described by the following equations:

$$\ddot{\phi}_{\substack{\text{avail.}\\(\epsilon \neq 0)}} = \left[\ddot{\phi}_{\substack{\text{avail.}\\(\epsilon = 0)}} \pm \frac{F_{op}\,\epsilon}{I_{xx}} \right]$$

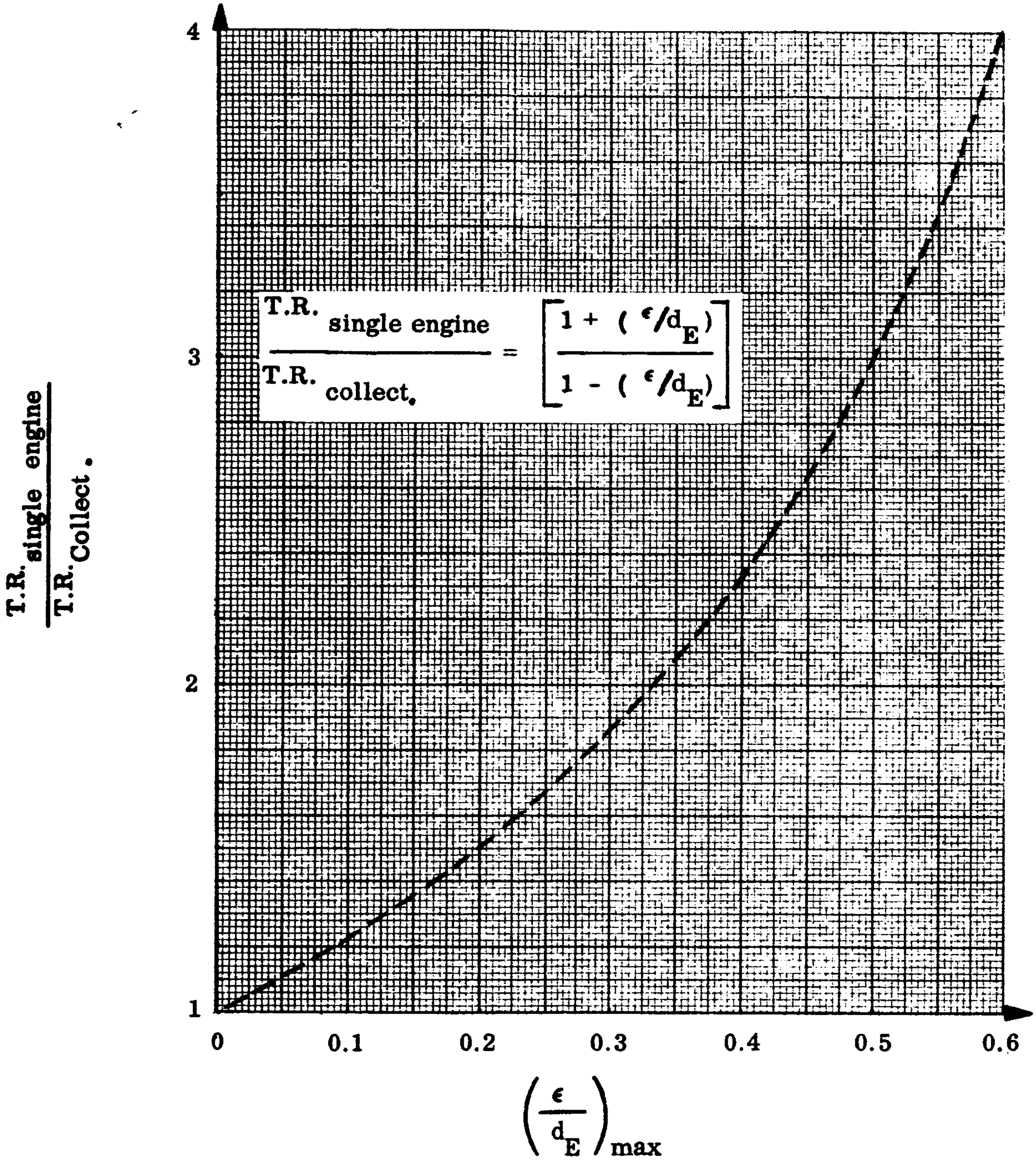


Figure 15. Engine Throttling Ratio Requirements as Influenced by the Overall Throttling Requirements and the c.g. shift Effects

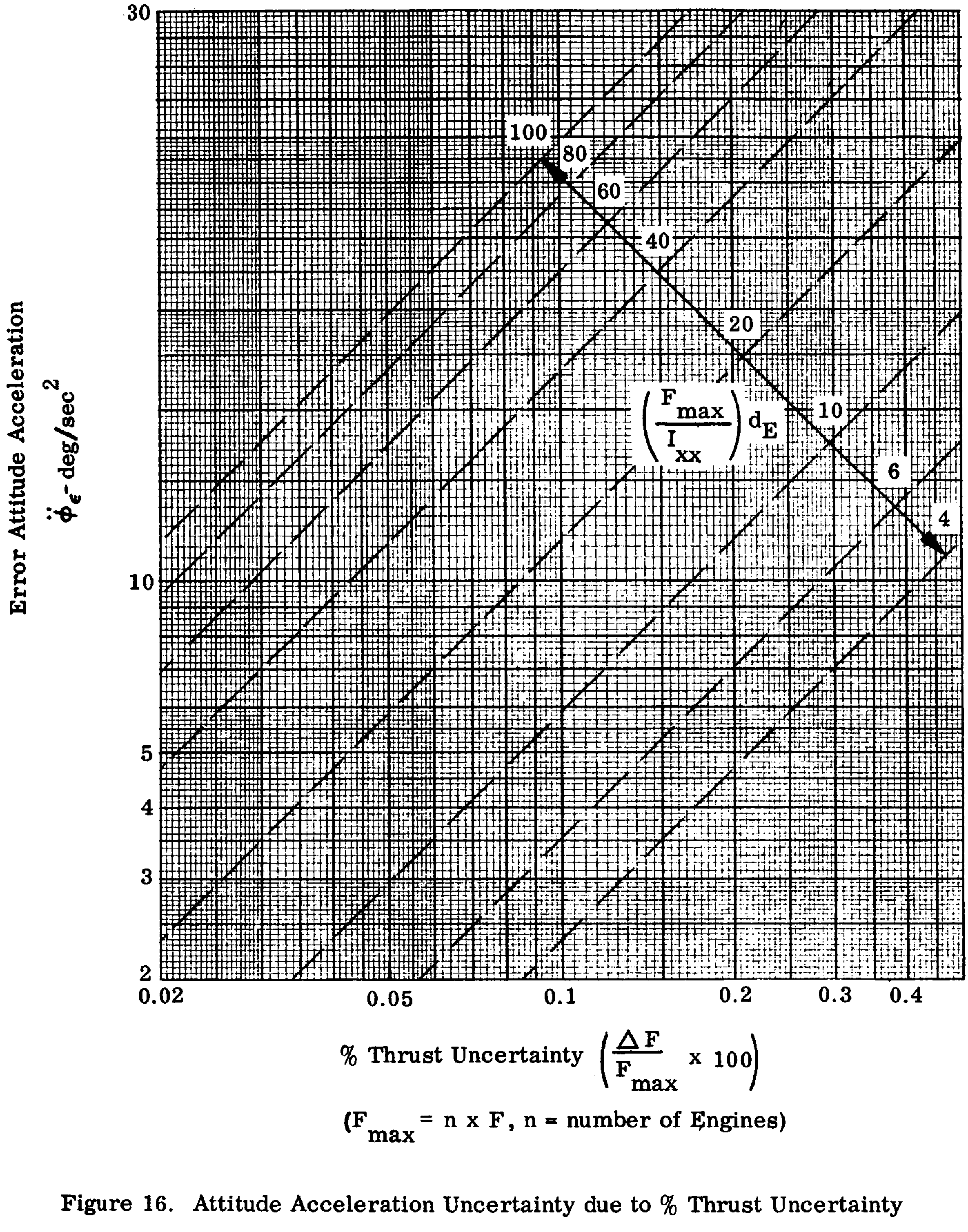


Figure 16. Attitude Acceleration Uncertainty due to % Thrust Uncertainty

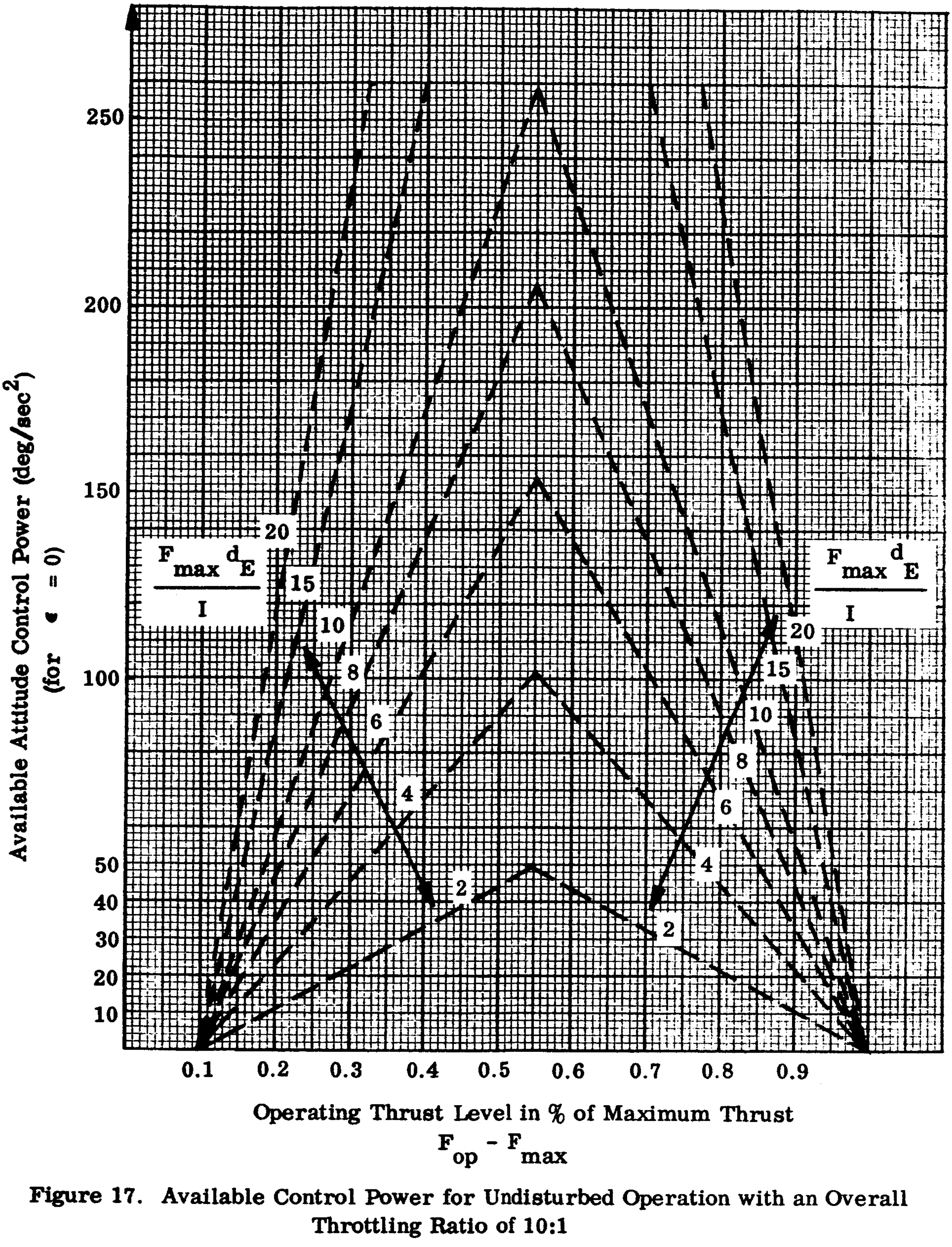


Figure 17. Available Control Power for Undisturbed Operation with an Overall Throttling Ratio of 10:1

$$= \left[\ddot{\phi}_{\substack{\text{avail.} \\ (\epsilon = 0)}} \pm \Delta \ddot{\phi} \right]$$

$$= \ddot{\phi}_{\substack{\text{avail.} \\ (\epsilon = 0)}} \left[1 \pm \frac{\Delta \ddot{\phi}}{\ddot{\phi}_{\substack{\text{avail.} \\ (\epsilon = 0)}}} \right]$$

Figure 18 shows the % change in control power (correction factor in the above equation) in terms of the % collective thrust and the ratio $\dfrac{\Delta \ddot{\phi}}{\ddot{\phi}_{\substack{\text{avail.} \\ (\epsilon = 0)}}}$. Figures 17 and 18 together may be used in determining the available control power for a given vehicle configuration and a given collective thrust program. This information can be used in screening suggested configuration and corresponding flight thrust programs.

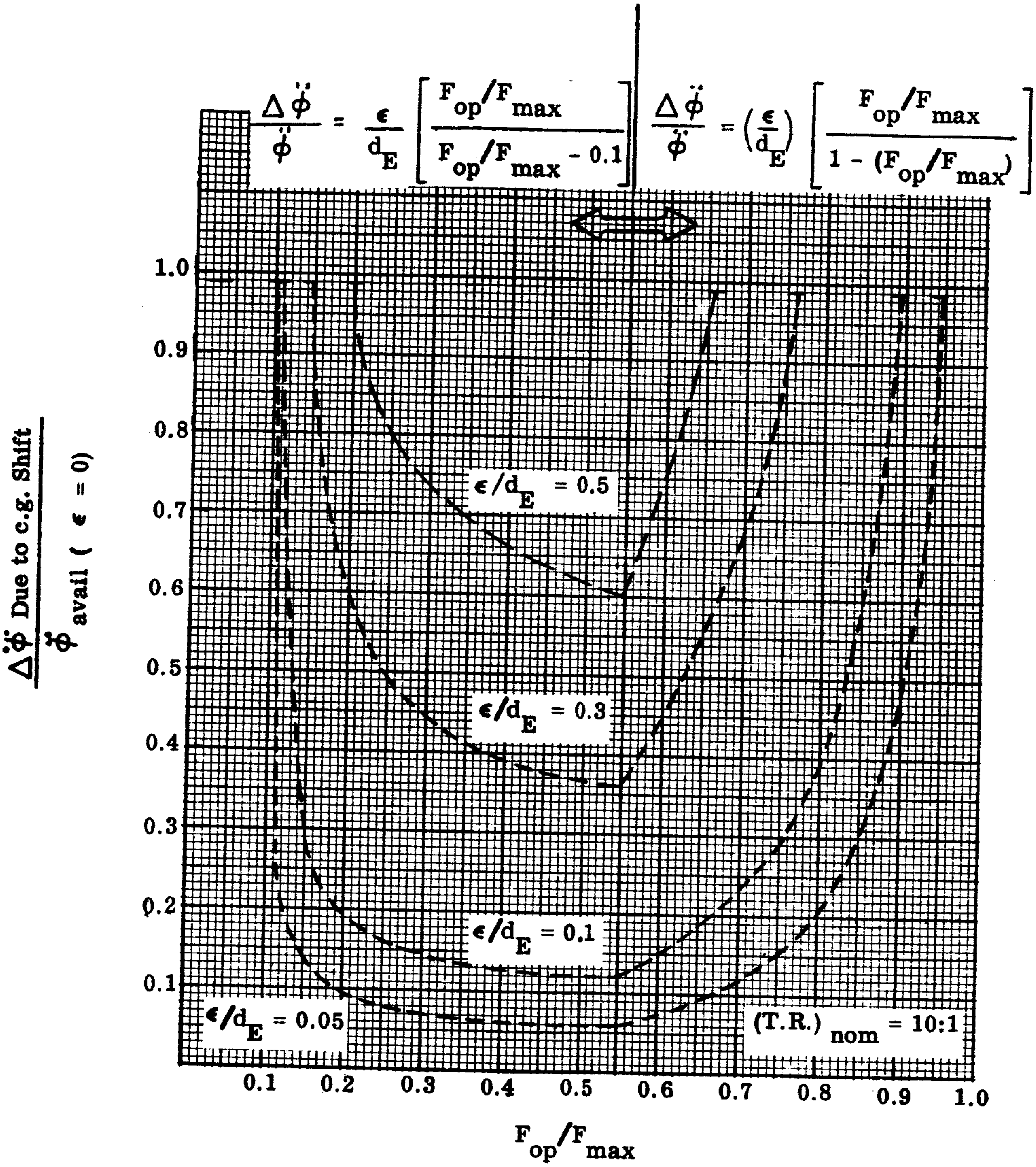


Figure 18. Relative Control Power Variation due to c.g. Shift Effects

D. SET OF RIGID, FIXED THRUST REACTION JETS

1. General

An attitude control system using a set of fixed thrust reaction jets is the most straight-forward approach for use on the majority of the proposed vehicle configurations. Advantages of reaction jet control are its simplicity of mechanization and operation, the capability of vehicle stabilization during main engine off periods (coasting phases) during a flight, and low hardware weight and propellant requirements. Previously discussed methods of main engine gimballing or main engine differential throttling used for gross c.g. shift compensation could be combined with a reaction jet control system used for compensating small residue c.g. shift effects and to provide commanded control moments. However, once a reaction jet system has been selected to provide attitude control it may be more convenient and economical to simply size the system to also be capable of handling gross c.g. shifts if their magnitude can be maintained within acceptable limits.

2. Number of Reaction Jet Thrusters

The minimum number of fixed thrusters in a 3-axes reaction control system is 6, but there may well be 8, 10, or 12 thrusters, or even more if redundancy and multi-thrust-level operations are considered. The choice of the suitable number of reaction jets for a given vehicle will be based on considerations like axes coupling, translational forces due to moments, hardware weight, redundancy, and the effects of all these on the overall vehicle performance in a given mission.

Use of 6 reaction jets - set of 2 per axis - results in simple moments being applied to the vehicle instead of pure couples. Application of a simple moment always causes a translational force to act on the vehicle, and if c.g. shifts exist a simple moment applied to one axis will also result in a coupled moment to another axis. If the reaction jet thrust level (f) is much smaller than the main engine (F) and the moment arm (ℓ) is much larger than the c.g. travel (ϵ), then the Δ V contribution from the attitude stabilization will be negligibly small compared to the Δ V from the main thrust. However, when these conditions are not met, considerable translational velocity may be imparted. It should be noted that the pitch and roll thrusters can be oriented so that their thrust adds to that of the main engine, thereby conserving propellant. However, when the reaction jet thrust is significant compared to the main engine thrust, the guidance system requirements may become more severe; for example, a velocity meter may be required during the boost phase to determine the cut off velocity for critical missions, where a simple timer and known main engine thrust would otherwise suffice. The severity of these effects and their influence on the overall vehicle controllability and flight performance must be evaluated for each particular configuration and mission. The cross-axis coupling and the moment-translation coupling effects described above may be eliminated by providing pure couples to obtain angular accelerations - set of 4 reaction jets per "uncoupled" axis. Thus a 3-axes reaction jet system which exhibits no coupling will have a minimum of 12 thrusters.

In the following analysis it was assumed, unless otherwise specified, that the vehicle's main thrust always acts along the vehicle body Z-axis, and that a minimum number of reaction jets - set of 2 per axis - is used. Figures 19 and 20 define the symbols used in this presentation.

3. Vehicle Stabilization

A misalignment of the total main engine thrust vector (F) and the actual c.g. of the vehicle will cause an error moment tending to pitch or roll the vehicle. This error moment ($F \cdot \epsilon$) will be countered by firing the appropriate reaction jet (or jets) such that the average moment produced by the reaction jet thrusting equals the error moment - this is illustrated in Figure 21.

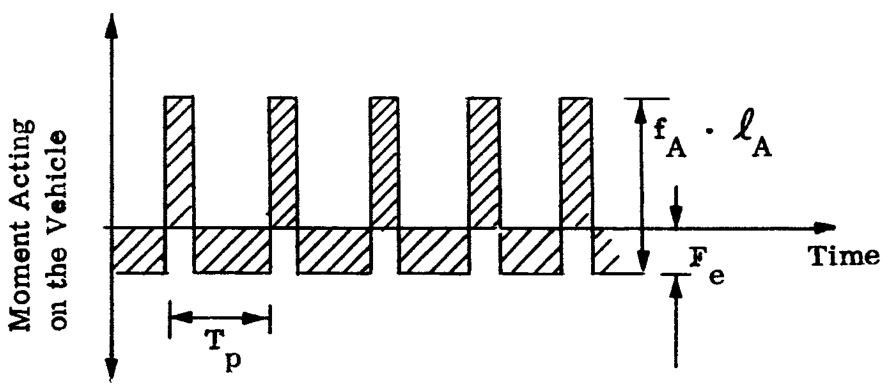


Figure 21. Error Moment Correction

The thrust level of the reaction jets required for stabilization (f_{St}) is determined by the available moment arm (ℓ), the maximum c.g. shift (ϵ_{max}), and the maximum thrust level of the main propulsion unit (F_{max}). Figure 22 shows this relationship.

C.g. shift compensation by means of reaction jet pulsing will always result in attitude limit cycling; the amplitude and frequency of the limit cycle is determined by the control system characteristics (gains, thresholds, lags, etc.).

4. Commanded Maneuvers

The available control power for a configuration using reaction jets is directly proportional to the thrust level (f) and the moment arm (ℓ), and is inversely proportional to the vehicle moment of inertia. As propellant is consumed the control power will increase.

Using the notation as defined in Figure 20 the following expressions can be written for the attitude control power for zero c.g. shifts.

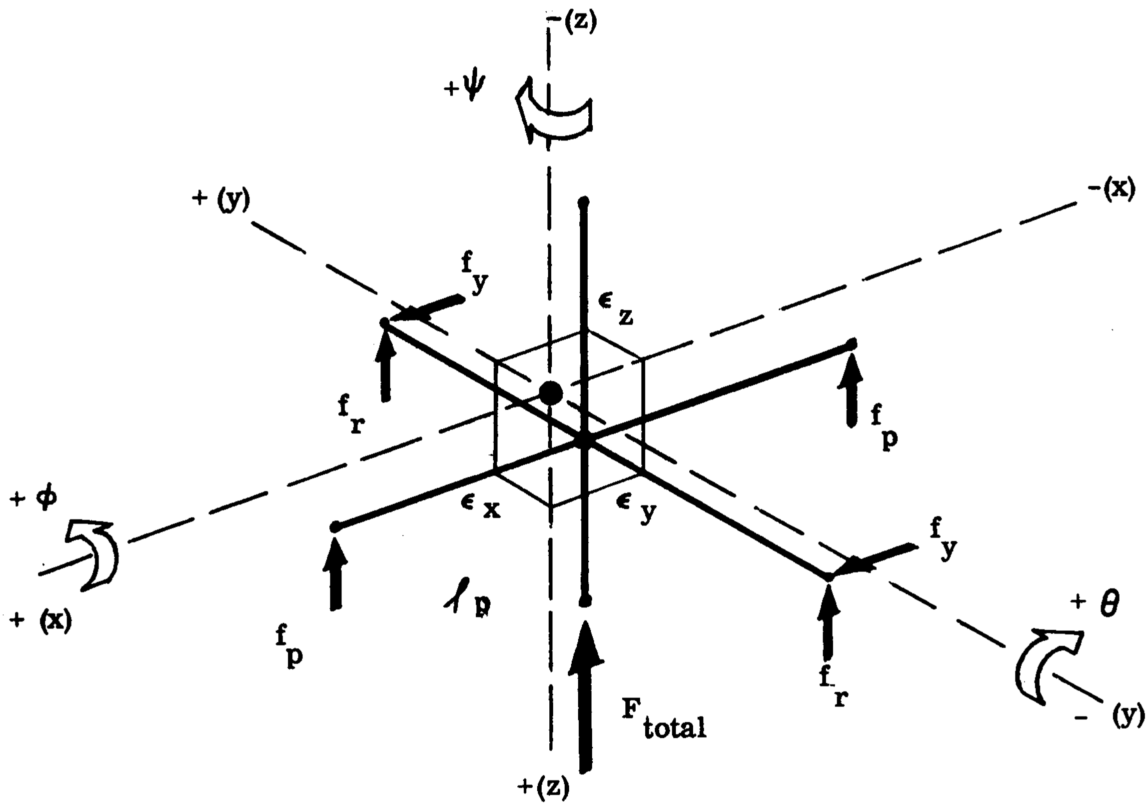


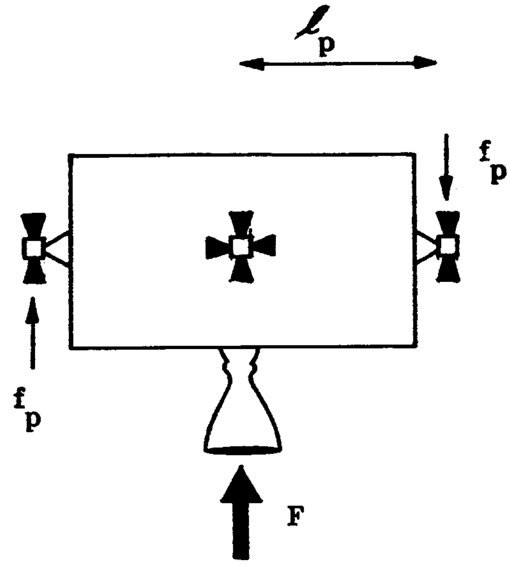
Figure 19. Symbol Definition

F	-	Main Engine Thrust	lb
f	-	Reaction Jet Thrust	lb
ℓ	-	Moment Arm	ft
I_{ii}	-	Moment of Inertia (About ii axis where ii = xx, yy or zz)	slug-ft^2
t	-	Time	sec
T_p	-	Period	sec
ϵ	-	c.g. Shift	ft

Subscripts:

p,r,y	-	Refers to Pitch, Roll, Yaw
x,y,z	-	Refers to Vehcile Body Axis
St	-	Refers to Stabilization
LC	-	Refers to Control

Set of 12 Reaction Jets



Set of 6 Reaction Jets

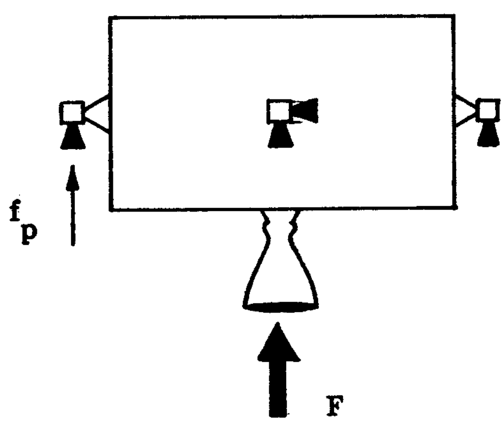


Figure 20. Reaction Jet Arrangement

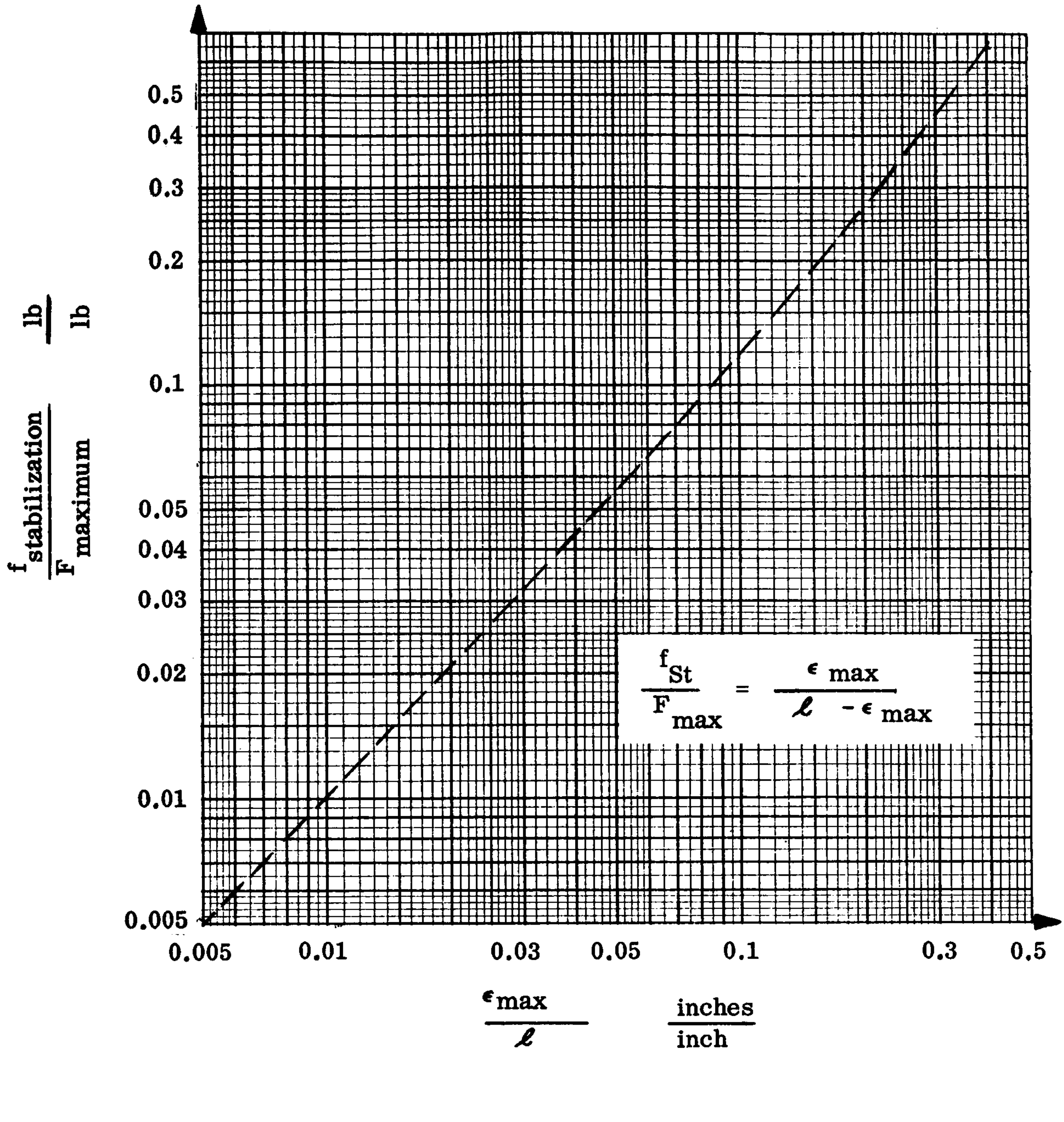


Figure 22. Reaction Jet Thrust Level Requirements for c.g. Shift Compensation

Pitch: $\ddot{\theta}_c = \dfrac{f_p \ell_p}{I_{yy}}$ (rad/sec^2)

Roll: $\ddot{\phi}_c = \dfrac{f_r \ell_r}{I_{xx}}$

Yaw: $\ddot{\psi}_c = \dfrac{f_y \ell_y}{I_{zz}}$

Figure 23 shows this relationship in terms of general parameters. This figure may be used in determining the required thrust levels in all three axes for a given vehicle configuration and specified control power levels.

If the requirements for attitude maneuvers are not in terms of control power but rather specify the time in which a particular turning maneuver is to be executed than Figure 24 can be used to determine the required reaction jet thrust for any axis in a given configuration (known values of ℓ and the corresponding I for the three axes). The total reaction jet thrust levels (f_p, f_r and f_y) are determined from the stabilization requirements and the control requirements as shown in the following general expression.

$$f = f_{stab.} + f_c$$

For a given vehicle configuration the three values of f for pitch, roll, and yaw are determined from Figure 22 and from either Figure 23 or Figure 24.

The moment requirements for pitch and roll frequently will be similar enough to permit use of identical thrust levels but the yaw requirements may be quite different ($f_{stab.}$ for yaw $\simeq$ zero).

5. Asymmetry of Attitude Moments Due to c.g. Shifts

For a vehicle as defined in Figure 20 the following general expressions for the attitude acceleration can be written.

Pitch: $$\ddot{\theta}\ (\text{rad/sec}^2) = \frac{f_p \ell_p}{I_{yy}} \pm \frac{(F + f_r + f_p)\,\epsilon_x - f_y\,\epsilon_z}{I_{yy}} = \underset{(\epsilon = 0)}{\ddot{\theta}_c} \pm \Delta\ddot{\theta}$$

Roll: $$\ddot{\phi} = \frac{f_r \ell_r}{I_{xx}} \pm \frac{(F + f_r + f_p)\,\epsilon_y}{I_{xx}} = \underset{(\epsilon = 0)}{\ddot{\phi}_c} \pm \Delta\ddot{\phi}$$

Yaw: $$\ddot{\psi} = \frac{f_y \ell_y}{I_{zz}} \pm \frac{f_y \epsilon_y}{I_{zz}} = \underset{(\epsilon = 0)}{\ddot{\psi}_c} \pm \Delta\ddot{\psi}$$

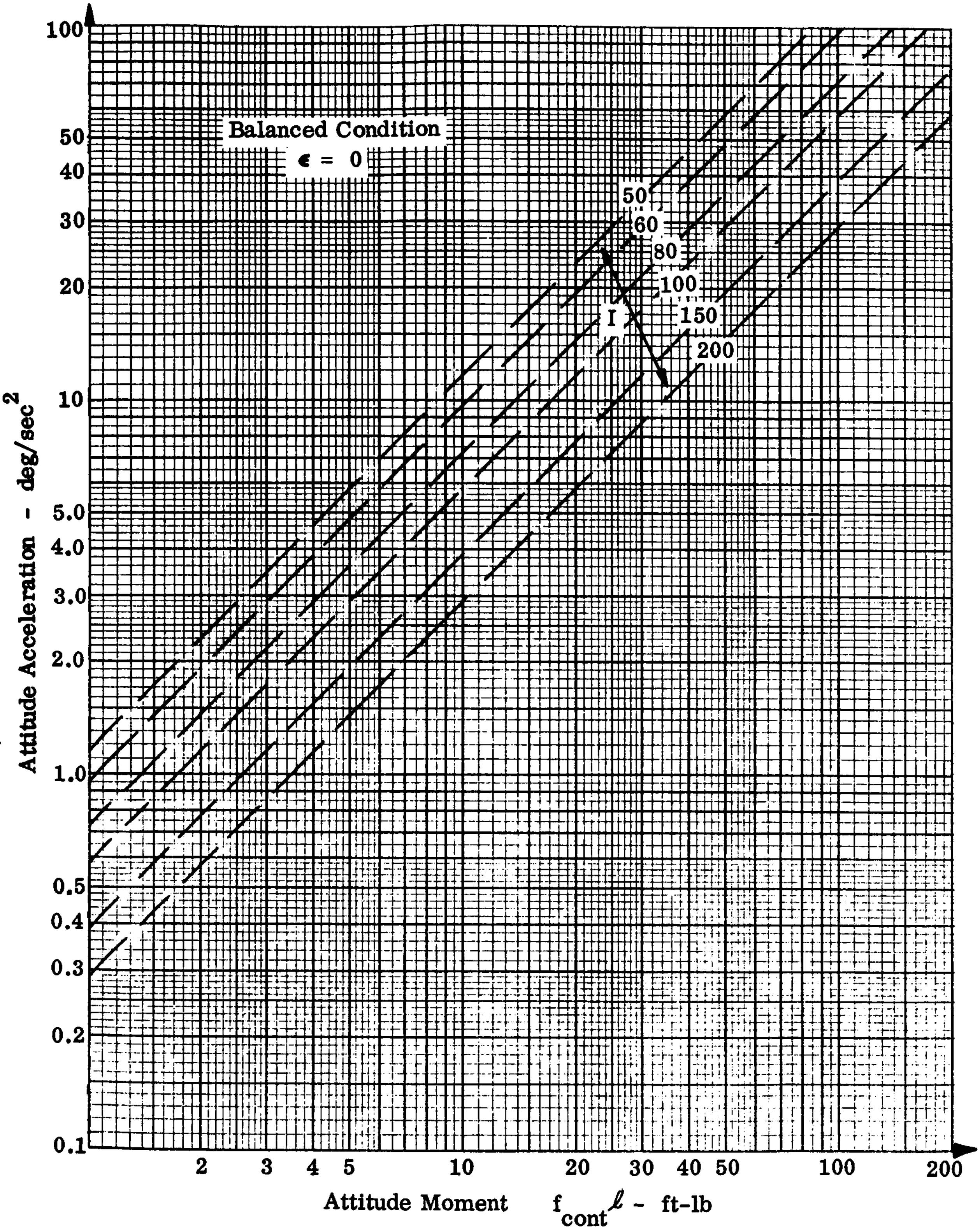


Figure 23. Reaction Jet Thrust Level Requirements for Commanded Attitude Acceleration

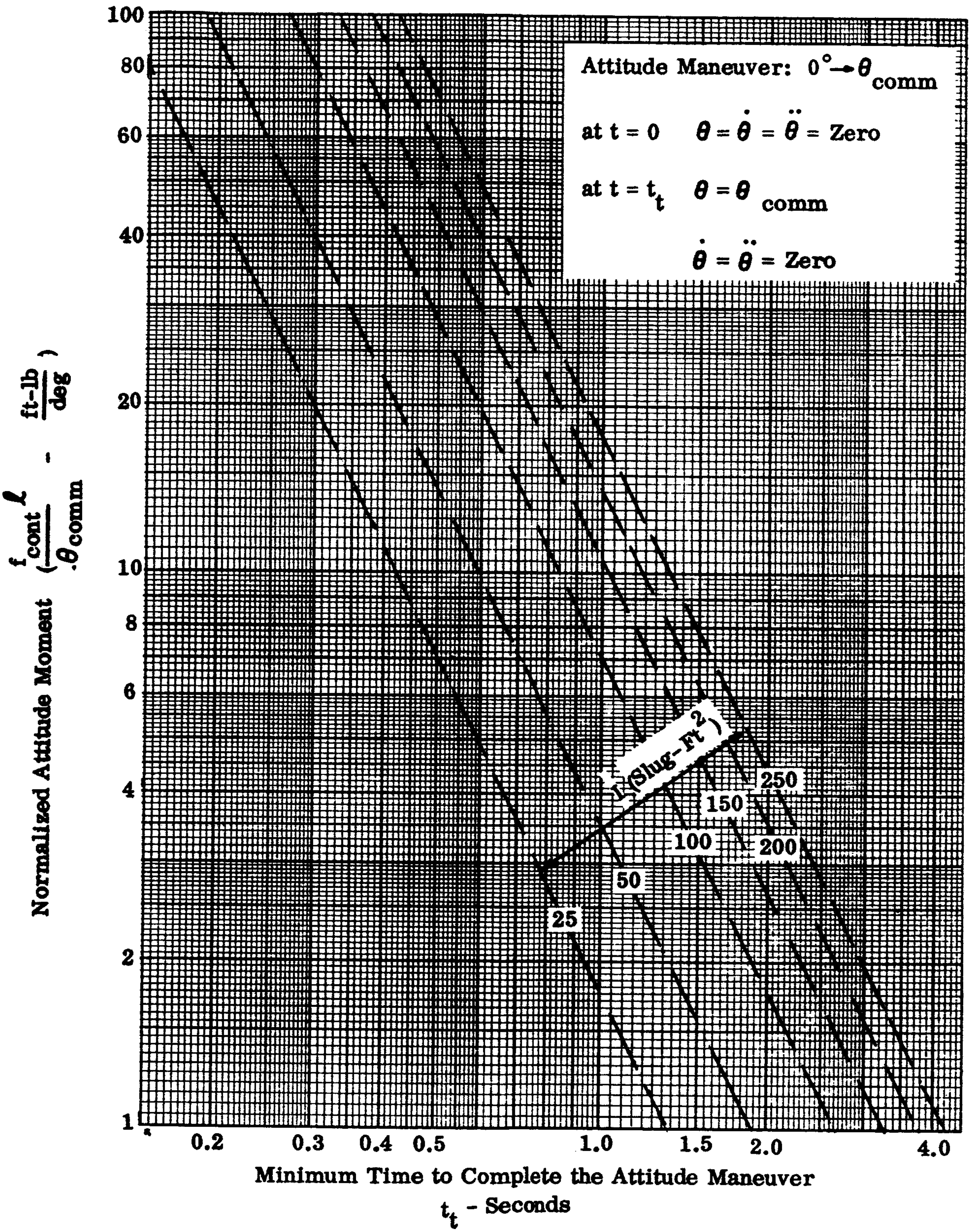


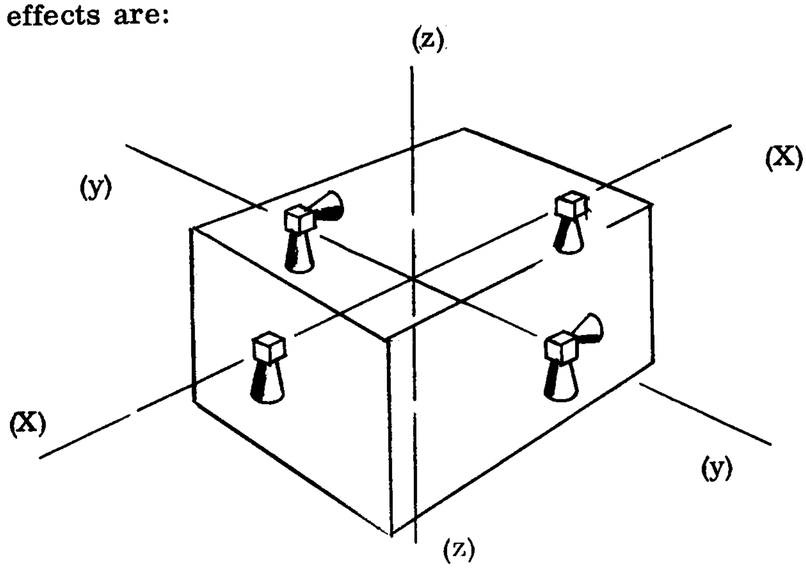
Figure 24. Reaction Jet Thrust Level Requirements for Commanded Attitude Trim

In these equations the first term represents the undisturbed attitude acceleration ($\ddot{\theta}_c$, $\ddot{\phi}_c$, and $\ddot{\psi}_c$) and the second terms represent the unbalance caused by c.g. shifts ($\Delta \ddot{\theta}$, $\Delta \ddot{\phi}$, and $\Delta \ddot{\psi}$).

For a given vehicle configuration and specified levels of control power ($\ddot{\theta}$, $\ddot{\phi}$,) and acceleration uncertainty ($\Delta \ddot{\theta}$, $\Delta \ddot{\phi}$) Figure 25 can be used to size the reaction jet thrust level for pitch and roll. Figure 25 is based on the first two of the above presented equations with the simplification that only thrusting in one axis at a time is considered.

6. Cross-Coupling of the Attitude Moments

If simple control moments instead of pure control couples are used for attitude maneuvering that is, if two reaction jets are used instead of four per axis pitch-roll, roll-pitch, yaw-pitch coupling will be caused by c.g. shifts. The magnitude of these coupling effects are:



Pitch-Roll coupling due to side c.g. shift:

$$\ddot{\phi}_{\text{coupl.}} = \ddot{\theta}_{\text{comm.}} \left(\frac{\epsilon_y}{\ell_p}\right)\left(\frac{I_{yy}}{I_{xx}}\right)$$

Roll-Pitch coupling due to fore-aft c.g. shift:

$$\ddot{\theta}_{\text{coupl}} = \ddot{\phi}_{\text{comm}} \left(\frac{\epsilon_x}{\ell_r}\right)\left(\frac{I_{xx}}{I_{yy}}\right)$$

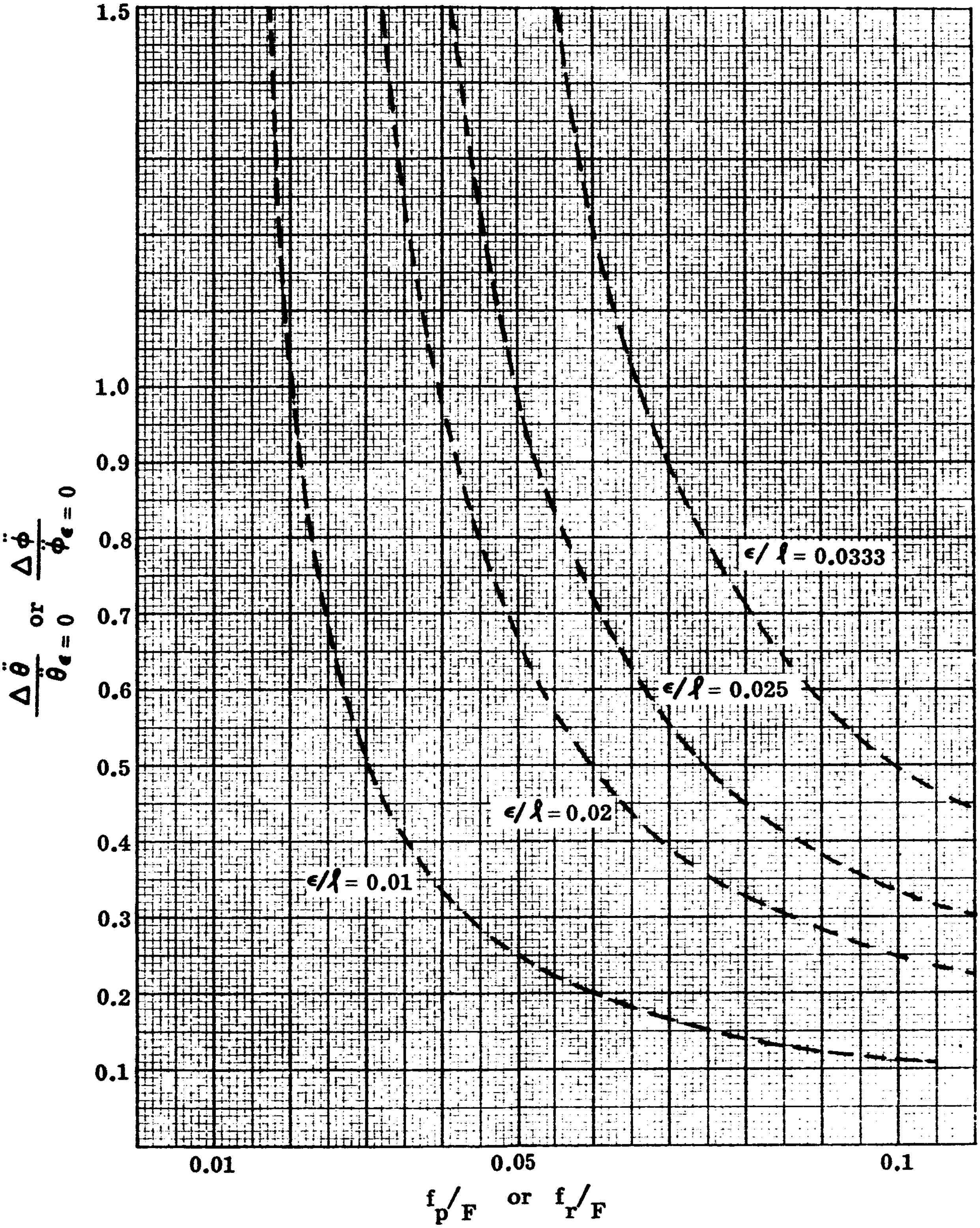


Figure 25. Total Reaction Jet Thrust Level Requirements in Terms of c.g. Shift Effects and Attitude Acceleration Requirements

Yaw-Pitch coupling due to vertical c.g. shift:

$$\ddot{\theta}_{\text{coupl.}} = \ddot{\psi}_{\text{comm.}} \left(\frac{\epsilon_z}{\ell_y}\right)\left(\frac{I_{zz}}{I_{yy}}\right)$$

Since, in general, the vertical c.g. shift due to propellant burn off (ϵ_z) is much larger than the horizontal c.g. shifts (ϵ_x and ϵ_y) there may occur rather large yaw-pitch coupling, as can be seen from the last of the above equations. Although there frequently is no need for large yaw moments, and consequently $\ddot{\psi}$ will likely be less than either $\ddot{\theta}$ or $\ddot{\phi}$, it may still be worthwhile to consider a pure couple system for the yaw axis. Thus a typical reaction jet system may utilize eight thrusters-two pitch, two roll, and four yaw.

E. MODIFICATIONS, SPECIAL CASES, AND RELATED MATERIAL

1. Mixed Vehicle Configurations

One attractive possibility for the vehicle attitude control and stabilization is a system which makes use of main engine gimballing or main engine differential throttling (para. II.B and C)) to counteract the gross error moments due to c.g. shifts, and where a set of low thrust reaction jets (para. II.D) accomplishes the vernier stabilization and the attitude maneuvering.

Some sort of a mixed system will in any case be required for missions involving ballistic flight trajectories - main engine off periods during which vehicle stabilization and (or) vehicle positioning is required. Questions of redundancy, operation under some failure condition, etc. may also point toward the need for a mixed system. Some care must be taken in considering a mixed configuration to account for the possible interferences between the different means of control and stabilization. Such interference may be minimized by use of a slow response control.

Establishing the requirements for the PPD backpack - the smallest, one man propulsion device similar to the Bell Rocket Belt applied to lunar or space environment operation - poses special problems due to the fact that in this case the man, being essentially the vehicle structure on which the engines, tanks, and other equipment are mounted, may be quite able to move parts of his body and consequently shift the c.g. and alter the moment of inertia.

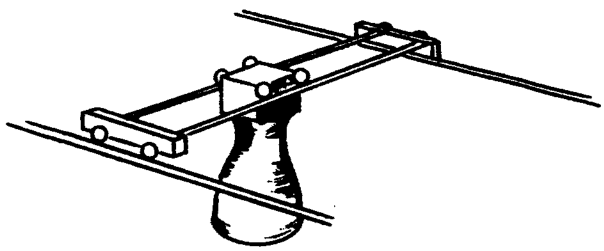
If in a first simple analysis it is assumed that the man either cannot move significantly due to the rigidity of his pressure suit, or does not choose to move, then the evaluation of the backpack device can proceed along the lines valid for the selection of vehicles with a rigid structure. In this case the three "pure" methods for producing moments, or a combination of these may be examined as to their applicability and suitability.

If the man's ability to move his body has to be taken into account, then the selection criteria, concepts, and relationships presented in the following sections are not applicable. The general performance of a flexible man with any backpack is influenced to a large degree by the motions of the man - orderly body motions to initiate altitude maneuvers and in response to altitude maneuvers as well as reflex actions. In this case the selection of a "best" or even satisfactory backpack arrangement - gimballable lift thrusters, set of reaction jets, etc. - can hardly be made on the basis of a rigid model.

2. Main Engine Translation

Compensation for a c.g. shift by means of a gimballed propulsion unit will require a change in the nominal attitude, as was shown in paragraph B. If, instead of gimballing the propulsion unit for thrust alignment, the propulsion unit is translated to achieve thrust - c.g. alignment then the nominal vehicle attitude will not change. (See Figure 26)

Engine Translation



Min. Range: $\epsilon\ x_{max}$

$\epsilon\ y_{max}$

(Vertical c.g. shift does not require engine adjustment)

Figure 26. Main Thruster Z-Axis Translation

Positioning of the propulsion unit is made either manually by the pilot in response to some attitude acceleration information, or positioning is made by means of servomotors in response to a c.g. control error signal.

The requirements for hardware and power, weight and complexity for propulsion unit translation will be of the same order as the requirements for a gimballed propulsion unit.

3. "Pseudo Throttling" by Means of Main Engine Pulsing

Attitude control and stabilization by on-off pulsing of the main engines is similar to that obtained with separate reaction jets - Section II.C. treats this method in detail.

Attitude stabilization by means of main engine pulsing - differing pulsewidth for the different engines corresponding to the error moment caused by the c.g. shift - is a possible but not a likely solution to the problem. To obtain a reasonable accuracy - steady state limit cycle amplitude of the order of a few degrees - would require a rather high pulse frequency (ω > 5 cycles per second), which in turn would result in a low effective I_{sp}.

Main engine pulsing for attitude control and stabilization would certainly require a rather sophisticated pulsewidth modulation system to achieve collective thrust modulation, attitude maneuvering, and minimization of limit cycles and coupling effects.

The performance of a system using main engine pulsing for the collective thrust modulation only compares favorably to the performance of a similar system using throttled engines. A comparison of the specific impulse values for the throttled and the pulsed operation indicated a rather substantial advantage of the pulsed thrust modulation. Figure 27 shows the ratio of the I_{sp} values versus the collective throttle ratio and the steady state altitude variation, and from the curves it is seen that the I_{sp} (pulsed) is substantially greater than the I_{sp} (throttled) over most of the throttling range.

Figures 28 and 29 illustrate the pulsing operations in terms of diagrams, phase plane trajectories, and general equations.

4. Prelaunch Balancing

At lift-off, the thrust of the vehicle's main propulsion unit may be such that the thrust-to-weight ratio of the vehicle is substantially larger than unity. Any significant misalignment of the thrust vector from the actual c.g. under these conditions will result in a large initial error moment. Although this error moment will immediately be counteracted by the vehicle's stabilization system, there may arise a condition where the resulting transient attitude error is very large ($\theta_{error} \geq 90° \rightarrow$ crash).

An initial vehicle unbalance may result from conditions such as difference in weight between pilot and passenger, relative position of each man in his seat, payload variations, incomplete propellant load, main engine misalignment, and deflection in vehicle structure. To protect against initial attitude errors would require large attitude control power, very likely much larger than that required for normal flight operation. The level of control power is generally proportional to the hardware weight and the power requirements of the system providing this control power - large gimbal range and rate, large thrust reaction jets, etc. The obvious desirability of a minimum weight attitude control system, sized according to normal flight disturbances and control power requirement, indicates the need for some prelaunch vehicle balancing to reduce any large initial c.g. uncertainties to within some small residue.

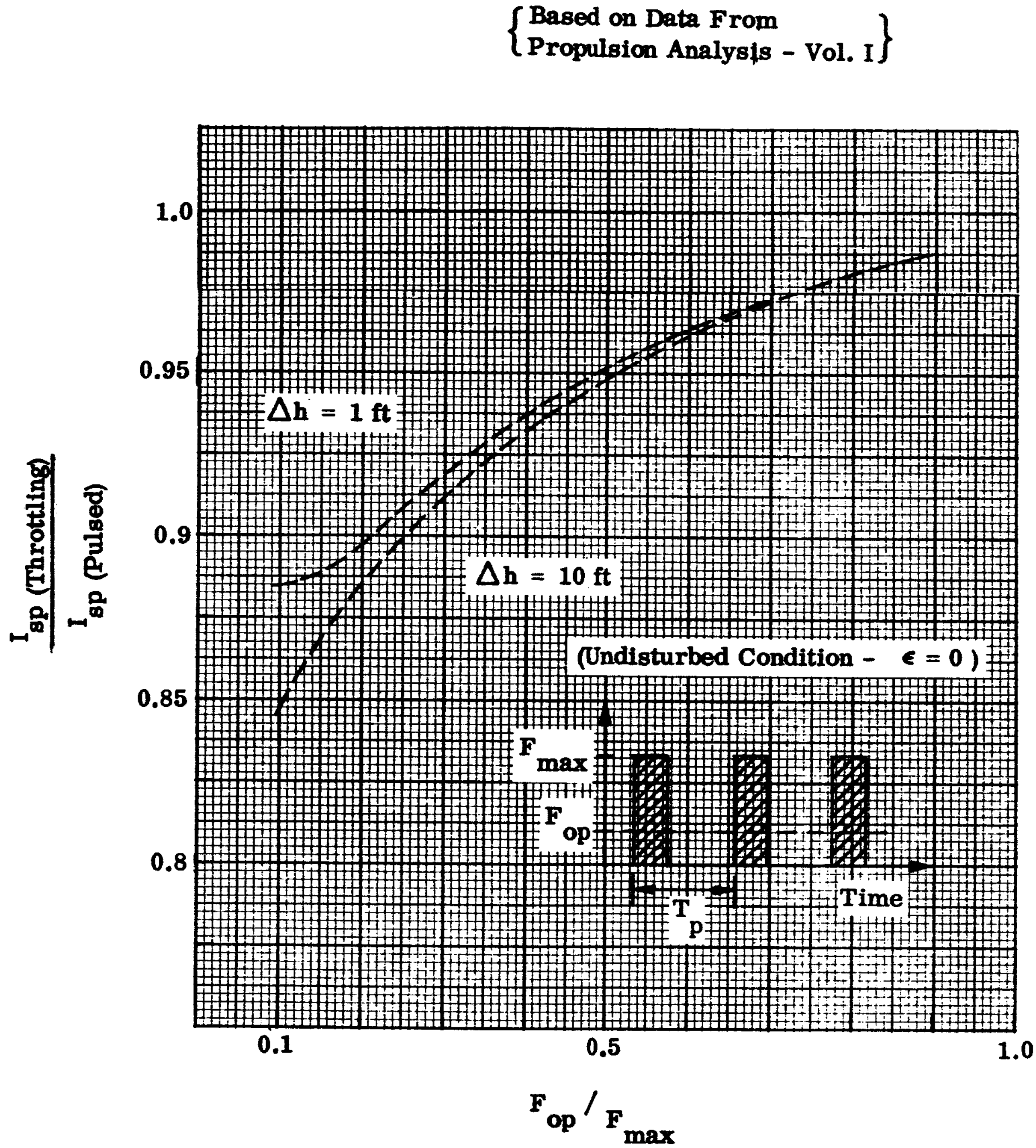
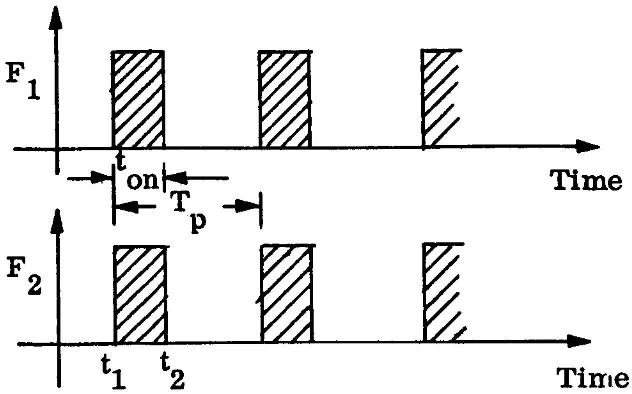
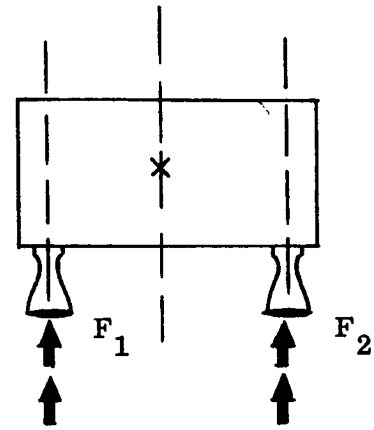
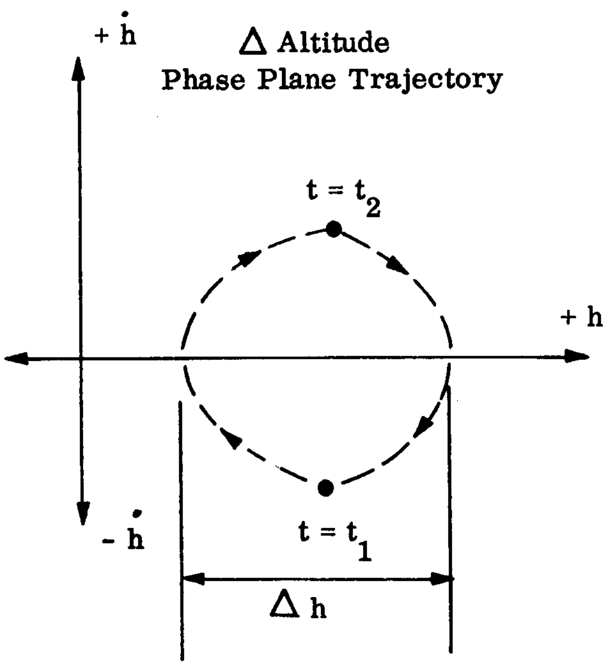


Figure 27. Propellant Efficiency Comparison for Pulsed versus Throttled Main Engine Operation

Undisturbed Condition ($\epsilon = 0$)







Total Impulse Per Cycle

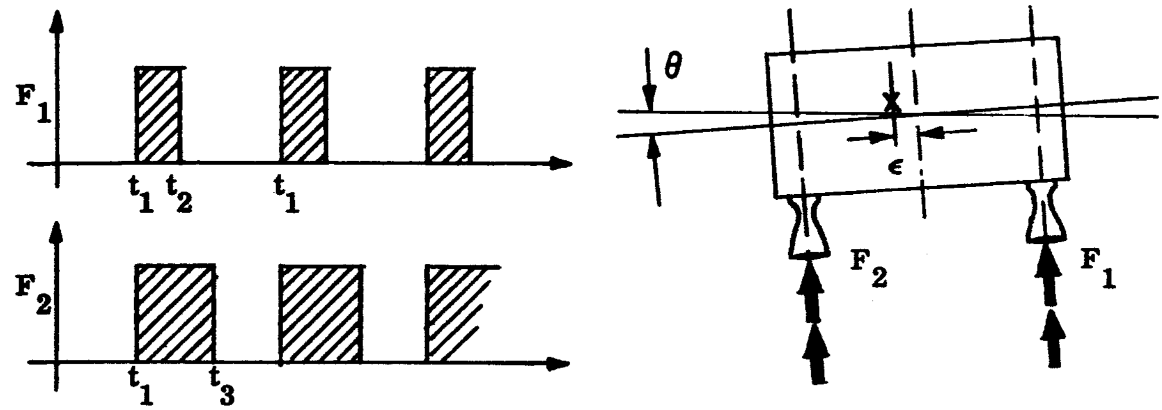
$$I_{tot} = (F_1 + F_2)(t_2 - t_1)$$

Altitude Variation

$$\Delta h = \frac{g_L}{8}\left(1 - \frac{g_L W}{g F_{max}}\right)(T_p)^2$$

$$= \frac{g_L}{8}\left[1 - \frac{\bar{F}_{op}}{F_{max}}\right](T_p)^2$$

Figure 28. Main Engine Pulsing for Collective Thrust Modulation



Total Impulse Per Limit Cycle Period

Translational Impulse	$I_T = F_1\,(t_2 - t_1) + F_2\,(t_3 - t_1) = I_T(\epsilon = 0)$
Angular Impulse	$F_1\,(t_2 - t_1)\,(d_E + \epsilon) - F_2\,(t_3 - t_1)\,(d_E - \epsilon) = 0$

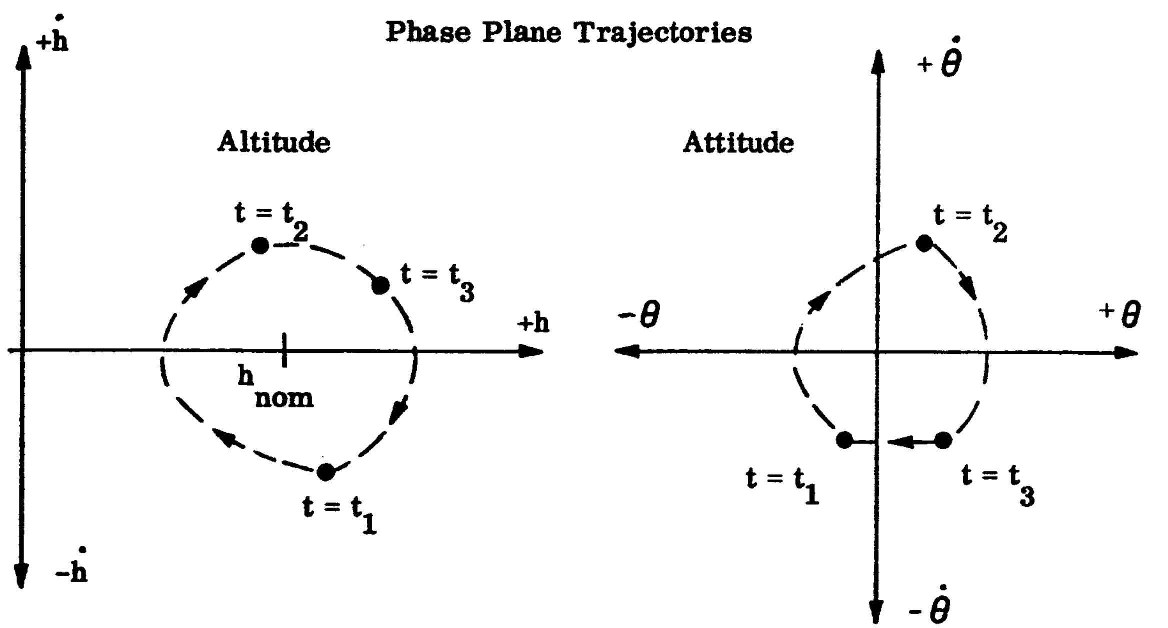


Figure 29. Steady State Pulsing With $\epsilon \neq 0$

The actual c.g. location is only known within some uncertainty ($\pm \epsilon$); the actual position and direction of the engine (or engines) is known only to within some uncertainty ($\pm \Delta d_E$, $\pm \Delta d_G$, $\pm \Delta \delta_G$). If the launch is initiated under these conditions there may result large error moments on the vehicle at lift-off requiring a large and fast acting attitude control system.

To avoid this undesirable initial unbalance the proper adjustments and alignments on the vehicle and engine can always be made such that the total thrust vector will be closely aligned with the actual c.g.

(1) Alignment of the c.g. with the actual Thrust Vector.

(a) Shifting of the load or some portion of the load (including shifting of the pilot) such as to align the c.g. with the thrust vector.

(2) Alignment of the Thrust Vector with the Actual c.g.

(a) Gimballing of one or more of the main engines.

(b) Differential throttling in a multiple engine configuration.

(c) Pulsing of the fixed thrust (attitude control) reaction jets.

The alignment may be executed (one or any combination of the four listed methods) in a simple manual - pilot executed and controlled procedure, or it may be done in a closed loop automatic c.g. seeking system.

The lift engine (or engines) are aligned such that the total thrust vector would be in the exact desired direction and at the same time goes through the vehicle c.g. - thus the vehicle as well as the engines are properly aligned. It is assumed that the desired launch direction is always along the local vertical and any attitude changes will be executed after lift-off. (See Figure 30)

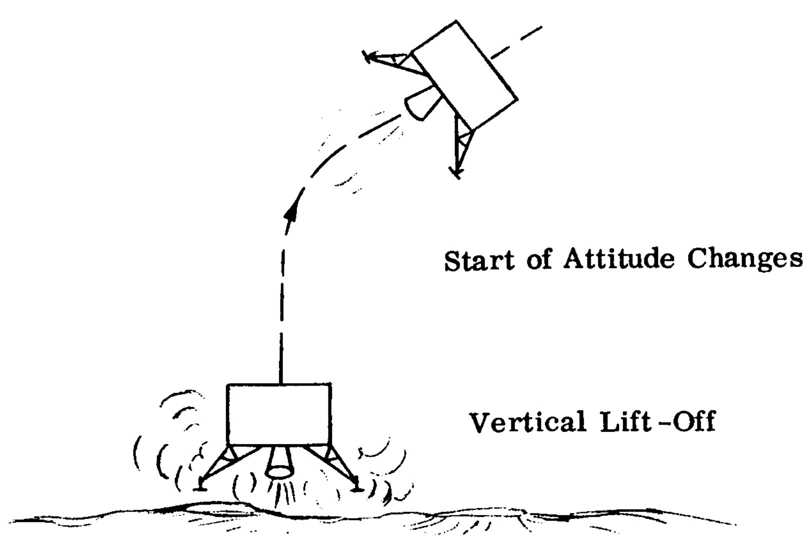


Figure 30. Lift-off and Attitude Change

Assuming vertical lift-off fixes the direction of the main engine (engines), and the remaining task is to align the vehicle c.g. directly onto the thrust axis of the main engine (engines). Three cases must be considered:

(1) Main Engine Gimballed

(2) Main Engines Rigid and Throttleable

(3) Main Engines Rigid and Fixed Thrust Reaction Jet-System

a. Main Propulsion Unit Gimballable

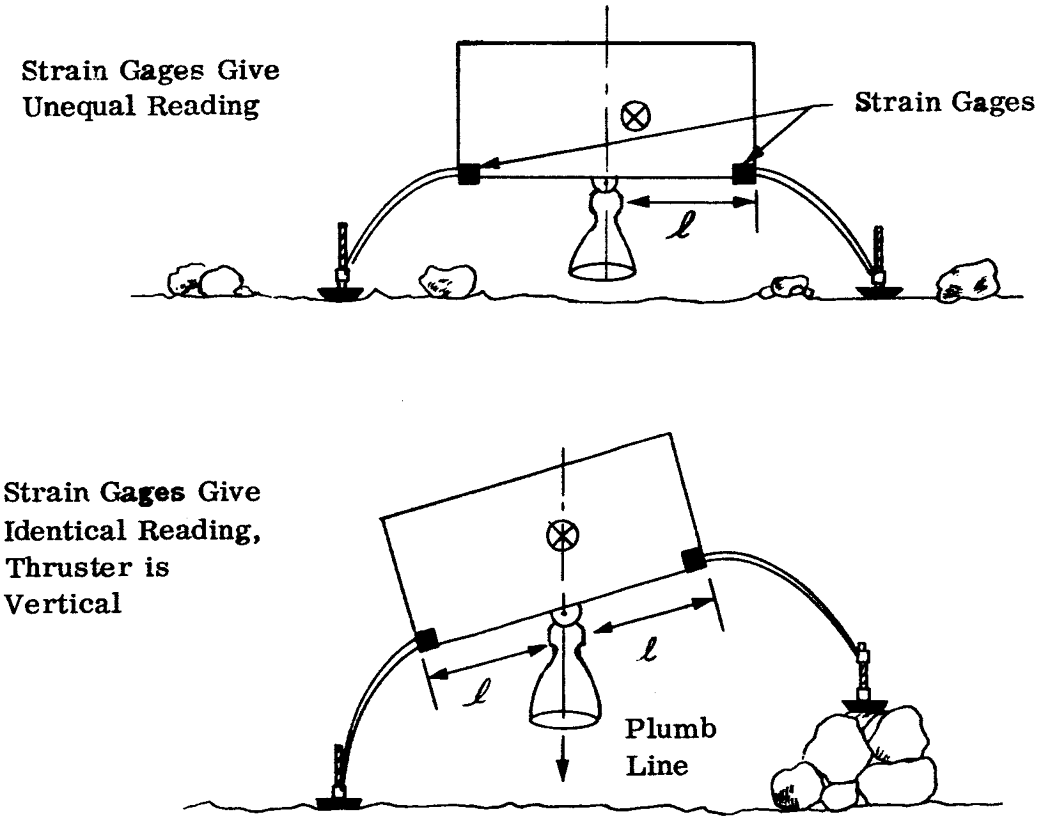


Figure 31. Main Propulsion Unit Configuration

Figure 31 illustrates one simple alignment procedure. Load indicating devices (strain gages, springs, etc.) located in the plane of the gimbal point, at a known distance from this gimbal point, and measuring parallel forces (forces parallel to the vehicle z-axis would be one satisfactory solution) are used to indicate the proper vertical alignment of the engine and the c.g.

The accuracy of the alignment depends on the accuracy with which the strain gage readings can be obtained and the accuracy of the dimensions ($\ell \pm \Delta \ell$).

b. Multiple Engine Configuration (Rigidly Mounted and Throttleable) (See Figure 32)

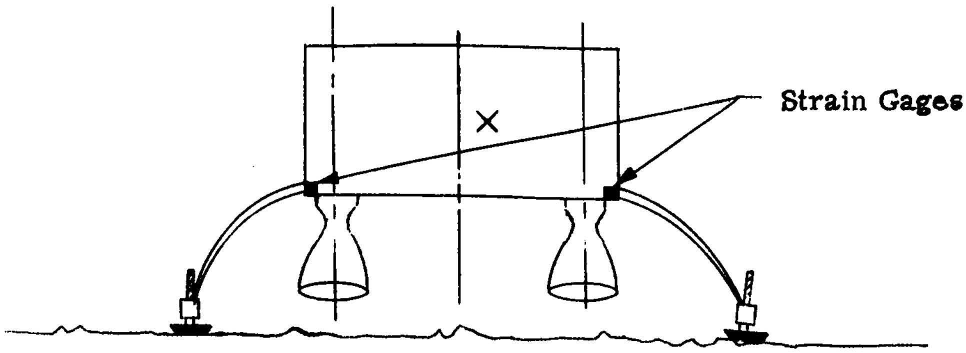


Figure 32. Multiple Engine Configuration

The vehicle is adjusted such that the engines are vertical, using a plumb line or a bubble level. Strain gages located at appropriate points indicate the horizontal location of the actual c.g. and thus indicate the required differential throttle setting for zero moment. Because this is not a nulling-type of adjustment, its accuracy depends on the accuracy of the strain gage readings, the dimensions, the engine position and the predictability of throttle setting.

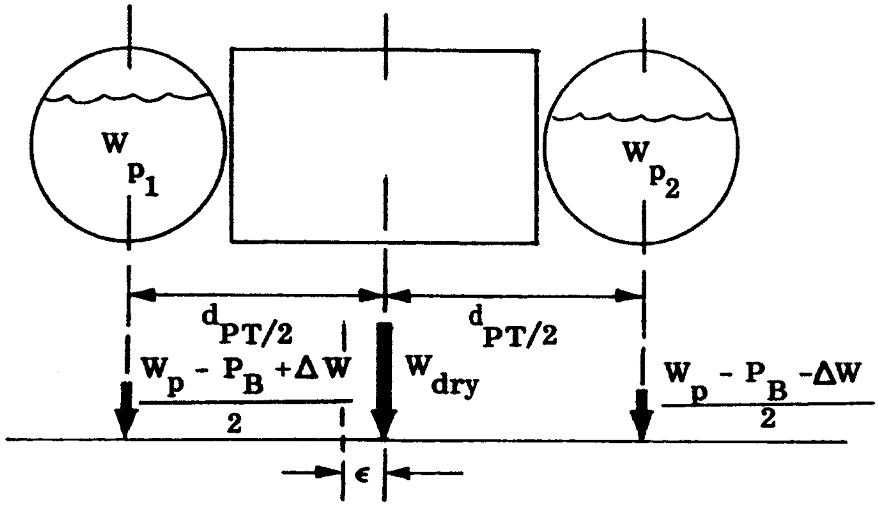
The method described below may also be applied to this case.

c. Rigidly Mounted, Fixed Thrust Propulsion Unit

Essentially same as b above, except that load shifting must be used to bring the strain gage readings into agreement - shift of the actual c.g. to coincide with the theoretical c.g. (Vertical c.g. shifts cause no error moments).

5. Normal c.g. Shift Effects Due to Propellant Burn Off

Although it can be assumed that at lift-off, the vehicle is balanced to within some very small residue ϵ_r c.g. shifts will occur during the vehicle's flight caused by propellant burn off and unequal tank drainage. All vehicle configurations, except the backpack unit, have a symmetrical tank arrangement consisting of two balanced sets of paired tanks for the oxidizer and the fuel. Utilizing these symmetrical tank arrangements restricts the c.g. travel due to normal tank drainage to the vertical axis. However, unequal tank drainage will produce small horizontal c.g. shift.



W_{dry}	=	Burn Out Weight
W_p	=	Total Propellant Weight (at Launch)
P_B	=	Weight of Propellant Drained (From Launch to a Given Time)
(% D.D.)	=	Percent Difference in Tank Drainage Rate
(% T.U.)	=	Percent Tank Unbalance Due to (% D.D.) = $\frac{1}{2}$ (% D.D.) $\frac{P_B}{W_p}$

Weight of Propellant Load to Any Given Time $(W_p - P_B) = W_{p_1} + W_{p_2}$

Tank Unbalance at Any Given Time $\Delta W = (W_B)\,(\%\ T.U.) = \frac{P_B}{2}\,(\%\ D.D.)$

$$\epsilon = \frac{d_{PT}}{4}\;\frac{P_B\ (\%\ D.D.)}{W_{dry} + W_p - P_B} \qquad \text{(c.g. Shift at a Given Time)}$$

$$\epsilon_{max} = \frac{d_{PT}}{2}\;\frac{(W_B)\ (\%\ T.U.)}{W_{dry}} \qquad \text{(c.g. Shift at Burnout)}$$

Figure 33. Horizontal c.g. Shift Due to Unequal Propellant Tank Drain

The tank geometry and the equations relating the c.g. travel in one horizontal axis to the particular vehicle parameters are shown in Figure 33.

The equations of Figure 33 only give the c.g. shift between a symmetrically arranged set of tanks (oxidizer or fuel). To obtain the c.g. shifts along the vehicle body axis, the equations of Figure 33 must be applied to both sets of tanks and the thus obtained c.g. shifts (ϵ_{oxid} and ϵ_{fuel}) must be converted to ϵ_x and ϵ_y. Figure 34 illustrates this procedure.

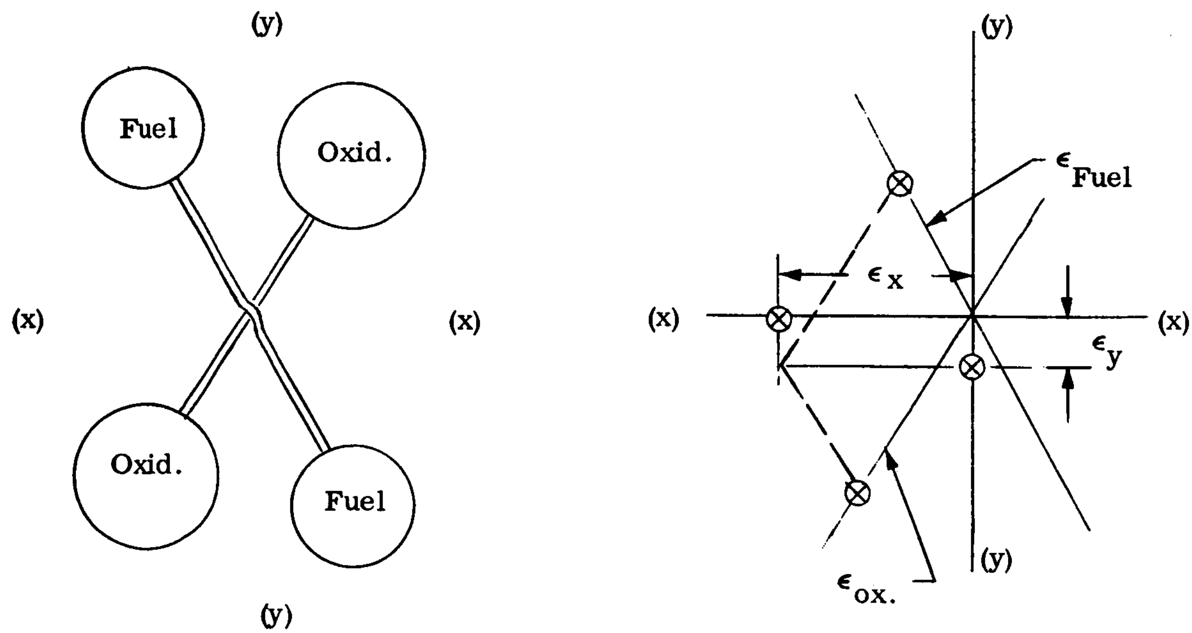


Figure 34. c.g. Shifts

F. SAMPLE APPLICATION OF THE SELECTION CRITERIA

1. Introductory Remarks

The various selection criteria presented in the preceding sections may be applied in evaluating the performance of any particular vehicle configuration. The performance indicators, thus obtained like control power and acceleration uncertainties for a vehicle using either gimballed main engines, differentially throttled engines, or reaction jet altitude controls can be compared, and based on this comparison some method of altitude control and stabilization most suitable for the particular vehicle may be selected.

If several equally satisfactory possibilities for obtaining control and stabilization emerge from the application of the various selection criteria, as may frequently be the case, then the final selection from these several possibilities may be based on considerations of overall system simplicity, reliability, suitability for backup operation, and general engineering judgment.

The various methods of altitude control and stabilization were evaluated on the basis of these performance requirements:

Capability to compensate the disturbing moments due to slowly changing c.g. shift in powered flight.

Adequate control power obtainable in all three axes during the entire flight was arbitrarily taken as 10 deg/sec^2.

Acceleration uncertainty (or resolution) in all three axes was equal to or less than an arbitrary value (0.5 deg/sec^2) for the entire duration of the flight.

2. Description of the Configuration used in this Sample Application of the Selection Criteria

Configuration 2.c. was chosen for this demonstration; it is a 2-man transportation device which exhibits enough flexibility to permit application of most of the selection criteria. Figure 93 and Tables XXVII and XXVIII present the pertinent system parameters.

a. Application of the Selection Criteria Presented in Paragraphs II.B-II.E.

(1) C.G. Shift Due to Unequal Tank Drainage - Application of Section VII

It is assumed that for both sets of tanks the flow rate for one tank differs by 2% from the flow rate for the opposing tank, and consequently the weight unbalance between a set of tanks will be equal to 1% of the total propellant drained from this set of tanks.

(% D.D.) = 2%   (% T.U.) = 1%

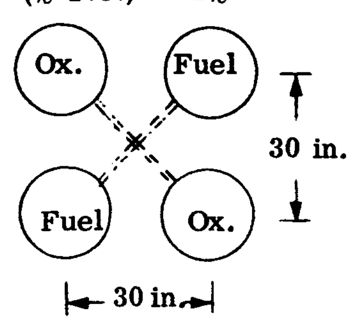
From Figure 93 and Tables XXVII

Burn Out Weight W_{dry} = 798.7

Ox. Weight W_{ox} = 455.0

Fuel Weight W_{fuel} = 285.0

Max. Vertical c.g. shift - ϵ_z = 8 in.



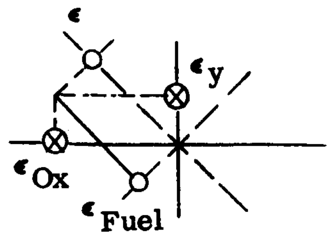
ϵ_z = 8 in.

Figure 33 - Section II.E.

$$\text{max} = \frac{d_{pt} \; W_p}{2 \, W_{dry}} \; (\% \text{ T.U.})$$

$$\text{ox} = \frac{(30\sqrt{2}) \; (455)}{2 \; 798.7} \; (0.01) = 0.12 \text{ in.}$$

$$\text{fuel} = \frac{(30 \;\; 2) \; (285)}{2 \; 798.7} \; (0.01) = 0.075 \text{ in.}$$



$\epsilon_x = 0.14$ in.

$\epsilon_y = 0.03$ in.

or

$$\left\{ \begin{array}{l} \epsilon_x = 0.03 \text{ in.} \\ \epsilon_y = 0.14 \text{ in} \end{array} \right\}$$

(2) Use of a Gimballed Main Engine - Application of Section II.B

For this example, the vehicle engine configuration consits of gimballed main thruster with a possible gimbal range of about ± 20 degrees, as indicated in Figure 93 .

From Figure 93 and Table XXVII

$F_{max} = 775$ lb Thrust

$F_{min} = 77.5$ lb Thrust

$d_G \simeq 11$ in.

$\ell_E \simeq 18$ in.

$$\left. \begin{array}{l} I_{max} = 163.1 \text{ slug-ft}^2 = I_{xx} \\ I_{min} = 82.9 \text{ slug-ft}^2 = I_{yy} \end{array} \right] \text{ for 200 lb man}$$

From Figure 2

Gimbal Range required for c.g. shift compensation.
At burnout the effective value of d_G will be

$$d^*_G \simeq d_G + \epsilon_z = 19 \text{ in.}$$

Using this value for d_G and the maximum horizontal c.g. shift valve ($\epsilon =$ 0.14 in.) the gimbal range can be found.

$$\delta_{G_{c.g.}} \simeq \pm 0.45 \text{ degrees}$$

From Figure 3

Gimbal angle resolution required to assure attitude acceleration uncertainties ≤ 0.5 deg/sec^2

$$\frac{F_{max}\, d^*_G}{I_{min}} = \frac{(775)\ (19)}{(82.9)\ (12)} = 14.8$$

Using the above indicated values the required resolution $\Delta\delta_G$ can be determined.

$\Delta\delta_G = 0.033$ degrees

From Section II.B.b.

Maximum rotation of the effective attitude reference axes with respect to the vehicle body axes is of the order of 0.5 degree.

Control Coupling Effect Negligible

From Figure 5

Gimbal range required for an attitude acceleration capability ≥ 10 deg/sec^2.

The most severe demands will exist during the final (landing) phase of the flight - low thrust operation at burn out.

$$\frac{F\ d_G}{I} = \frac{F_{min}\ d^*_G}{I_{min}} = \frac{(77.5)\ (19)}{(82.9)\ (12)} = 1.48$$

Using the above indicated values the gimbal range for command maneuvering can be determined.

$$\delta_G = \pm \left[6.8 + \delta_{G_{c.g.}}\right]$$

$= \pm$ 7.25 degrees

From Figures 7 and 8

Requirements for gimbal acceleration, gimbal saturation velocity, and gimbal range for executing a 40 degree turn in 4 seconds.

The most severe demands will again exist during the final phase of the flight.

$$\frac{F\ d_G}{I} = 1.48$$

$\delta_G \geq 41.7$ deg/sec^2

$\dot{\delta}_{G_{sat.}} \geq 13.5$ deg/sec^2

$\delta_G \geq 14.3$ deg

(3) Use of Differentially Throttled Main Engines - Application of Section II.C.

For this case it is assumed that the vehicle has 4 rigid, but individually throttleable main engines with a throttling ratio of 10:1 each, as indicated in Figure 93.

From Figure 93 and Table XXVII

$$F_{max} = 4 \times F_{max} = 772 \text{ lb thrust}$$

$$F_{max} = 193 \text{ lb thrust}$$

$$F_{min} = 19.3 \text{ lb thrust}$$

$$d_E = 7.5 \text{ in.}$$

$$I_{max} = 163.1 \text{ slug-ft}^2$$

$$I_{min} = 82.9 \text{ slug-ft}^2$$

Hover flight at burn out requires a collective thrust of about 133 lb

$$(\text{T.R.})_{collect} > \left(\frac{772}{133} : 1\right)$$

$$6:1 < (\text{T.R.})_{coll.} \leqslant 10:1$$

From Figure 14

Differential throttling ratio required for c.g. shift compensation.

$$\left({}^{\epsilon}/d_E\right)_{max} = \frac{\epsilon_{ox}}{d_E\sqrt{2}} = \frac{0.12}{7.5\sqrt{2}} = 0.0113$$

$$F_4 / F_1 \cong 0.96$$

Using the value and curve II in Figure 14 the throttling requirements for stabilization can be determined.

$$(\text{T.R.})_{c.g.} \simeq 1.05:1$$

Collective thrust difference due to c.g. shift compensation when operating at either high or low collective throttle.

Thrust Difference

$< 2\%$ of F_{max}

≈ 10 lb

From Figure 15

Reduction in overall throttle capability due to c.g. shift compensation.

$$(\text{T.R.})_{single\ E} = 10:1$$

$$(\text{T.R.})_{collective} \simeq \frac{1}{1.01} (\text{T.R.})_{single\ E} \simeq 9.9:1$$

From Figure 17

Thrust resolution required to assure attitude acceleration uncertainties $\leq$ 0.5 deg/sec^2.

The most severe requirements will exist during the final (landing) phase of the flight.

$$\frac{F_{max}\, d_E}{I_{min}} = \frac{(772)\ (7.5)}{(82.9)\ (12)} = 5.81$$

$\Delta F \leqslant 0.15\%$ of F_{max}

$\leqslant 0.57$ lb

From Figure 17

Available Control Power

The most severe requirements will exist at the initial (lift off) phase of the flight.

$$\frac{F_{max}\, d_E}{I_{max}} = \frac{(772)\ (7.5)}{(163.1)\ (12)} \approx 3$$

Maximum Control Power

at $F_{op} = 0.55\ F_{max}$

$(C.P.)_{max} \cong 75$ deg/sec^2

Range of operating thrust levels (F_{op}) for which the control power requirement of 10 deg/sec^2 is satisfied.

$0.16\ F_{max}\quad F_{op}\quad 0.94\ F_{max}$

(4) Use of Fired Thrust Reaction Jets - Application of Section II.D.

For this case it is assumed that the vehicle has a rigid main propulsion unit for lift and translation and set of fixed thrust reaction jets as indicated in Figure 93.

From Figure 93 and Tables XXVII

F_{max} = 775 lb

f_{pitch} = 10 lb

f_{roll} = 10 lb

f_{yaw} = 5 lb

ℓ_A = 40 inches

From Figure 22

Reaction jet thrust levels in pitch and roll required to compensate horizontal c.g. shift effects

$$\frac{\epsilon_{max}}{\ell_A} = \frac{0.14}{40} = 0.0035$$

$f_{St,} = 0.0035\ (F_{max})$

$= 2.71$ lb

(pitch and roll thrusters only)

From Figure 23

Reaction jet thrust levels to obtain control power of 10 deg/sec^2.

Lift-Off Conditions - Max. I, c.g. Shift = 0

Pitch - $f_p \cong$ 7.5 lb

Roll - $f_r \cong$ 9 lb

Yaw - $f_y \cong$ 8.4 lb

Required Thrust

$f_y \cong$ 8.4 lb

(based on $10^\circ/\text{sec}^2$)

Landing Conditions - Min. I, Max. c.g. Shifts

Pitch - $f_p \cong$ 4.5

Roll - $f_r \cong$ 5.4 + f $\simeq$ 2.7 lb

Yaw - $f_y \cong$ 4.5

$f_p \cong$ 7.5 lb

$f_r \cong$ 9 lb

For the ptich and roll axes the control power requirements will always be satisfied, but for yaw 10 deg/sec^2 acceleration capability may not always be achievable with 5 lb thrusters. In general roll and yaw maneuvering is not as important as is pitch maneuvering.

Available Thrust (Figure 93)

f_p = 10 lb

f_r = 10 lb

f_y = 5 lb

3. Comparison and Selection

Use of main engine gimballing or main engine differential throttling for stabilization and maneuvering would not eliminate the need for a set of low thrust reaction jets to be used for stabilization and maneuvering during main-engine-off periods. If the control power requirements during main-engine-off periods are anywhere near the requirements for powered flight (10 deg/sec^2) then the reaction jet thrust requirements are of the order of 5 to 10 lbs.

In addition to this, the accuracy requirements for gimballing or differential throttling are rather severe if an acceleration uncertainty of 0.5 deg/sec^2 is to be maintained under all conditions.

Required Gimbal Accuracy $$\frac{\Delta \delta_G}{\delta_G} \leq 0.45\%$$

Required Throttle Accuracy $$\frac{\Delta F}{F_{max}} \leq 0.15\%$$

Gimballing or differential throttling for attitude maneuvering is ruled out on the basis of these severe accuracy requirement indicated above.

Use of main engine gimballing on main engine differential throttling for vehicle stabilization - compensation of gross c.g. shift effects - would not result in any reduction of the required reaction jet thrust levels of 10 pounds for pitch and roll and 5 pounds for yaw.

Savings in propellant due to reduced need for corrective reaction jet pulsing may accrue with gimballing, but the magnitude of these possible savings and the consequent increase in the overall Δ V-potential is not likely to be significant.

The severe accuracy requirements for differential throttling (0.15% of F_{max} allowable thrust uncertainty) still apply and thus rule out this method. Gimballing for c,g. shift compensation would require a gimbal range of ±0.45 degrees with an accuracy of 10%. A simple, manually operated, gimbal mechanism would be workable, but only if the additional hardware weight of providing engine gimballing is less than the savings in propellant and thruster weight would this method be worth considering. This weight-saving-criteria may be relaxed if gimballing for an emergency backup operation is considered.

Since for the case at hand the advantages to be gained by adding gimballability for c.g. shift compensation are marginal the selected thruster configuration consists of a single, rigidly mounted, and throttleable main lift engine and a set of 6 attitude control reaction jets of 10 pounds thrust each for pitch and roll and 5 pounds thrust each for yaw.

SECTION III

PROPULSION SYSTEM

A. SUMMARY AND CONCLUSIONS

This section of the report consists of a summary of the propulsion analysis conducted to define the weight, performance, and envelope of propulsion systems using radiation cooled and ablative thrust chambers with $N_2O_4/0.50\ N_2H_4 + 0.50$ UDMH propellants. The thrust levels investigated from range 100 to 2500 lb and expansion area ratios from 20 to 120, at or near the optimum chamber pressure for this application. The $N_2O_4/0.50\ N_2H_4 + 0.50$ UDMH propellants are used to benefit from commonality with the current Apollo vehicle, and because they offer the best performance and technology levels of current storables. A radiation cooled thrust chamber operating at a chamber pressure of 80 psia and an equal volume mixture ratio of 1.6 is recommended as the most suitable system. The 80 psia chamber pressure resulted from an optimization analysis, the 1.6 mixture ratio offers the best specific impulse for the throttle range for systems using fuel barrier cooling and the radiation cooled thrust chamber provides thrust levels independent of its firing history (up to one hour cumulative duration) in a lightweight, reliable propulsion system.

Use of a solid propellant motor for the boost phase was also investigated to incorporate the simplicity advantages of the high performance solids into the escape mission. The results of this effort are described in Volume II (Confidential).

Primary study emphasis was given to the radiation cooled system. A detailed examination of the delivered performance over the required thrust range was required to permit comparison of multiengine configurations with single engine configurations without inducing a bias from scale effects which are significant in the 100 to 500 lb thrust range. Empirical data combined with the Bray analysis were used to minimize performance efforts and to provide the basis of the thrust chamber designs used to define the weight, envelope, and operating characteristics described herein. This system utilizes current technology to provide a minimum of complexity in mission simulation with a minimum of induced input errors over the complete range of interest. Other design options are identified and described for mission comparison purposes.

A survey of potential attitude control systems was conducted to define the weight and performance of the most suitable systems for this application. A system using propellants drawn from the primary propulsion system tanks was found to offer the lightest weight and highest mission flexibility. This approach does not compromise the design or operational characteristics of either the primary or attitude control systems for thrust levels above 10 lb per thrust chamber. Lower unit thrust levels are feasible, but incur modifications to the pressure regulation system, the propellant flow control, and different mixture ratios for the primary and ACS chambers. In addition, residual propellant considerations will require detailed predefinition of the duty cycles or additional propellant reserves for each mission.

B. SYSTEM ANALYSIS

1. Approach

The parametric data was generated by the definition of a suitable vehicle configuration and sizing of representative propulsion systems followed by the parametric variation of the major weight elements for a range of thrust and gross weight levels. Performance, weight, and envelope requirements were defined over the desired parametric range at the optimum pressures using point designs from the thrust chamber and weight groups.

The following ground rules were used for this program:

(1) A common baseline for all parametric data was used to maintain the significant differences between configurations which result from the mission simulations.

(2) System design philosophy

 (a) Minimum system complexity

 (b) Man rating of system required

 (c) Technology limited to state-of-the-art or minimum development risk where feasible.

(3) Define and establish weight and performance effects of design compromises.

2. Biliquid N_2O_4/50-50 System

A representative N_2O_4/50-50 propulsion system was sized to accomplish the specified missions (ΔV 2000 to 9000 fps) for a range of expected nonpropulsive weights to define the thrust level, gross weight, and firing durations required by the majority of the configurations to be investigated in this study. The nonpropulsive weight (W_{NP}) is defined, for the purposes of this study, as the total lunar surface lift-off weight (in earth pounds) less the weight of the primary propulsion system. W_{NP} includes the net payload, the astronaut, the vehicle structure and guidance equipment, and the attitude control system weights.

A schematic of the representative system is shown in Figure 35. The system consists of from 1 to 5 radiation cooled thrust chambers operating fixed thrust as shown by the solid lines or with up to a 10 to 1 throttling capability using the gas assist system shown dotted in Figure 35. The propellants are contained in spherical

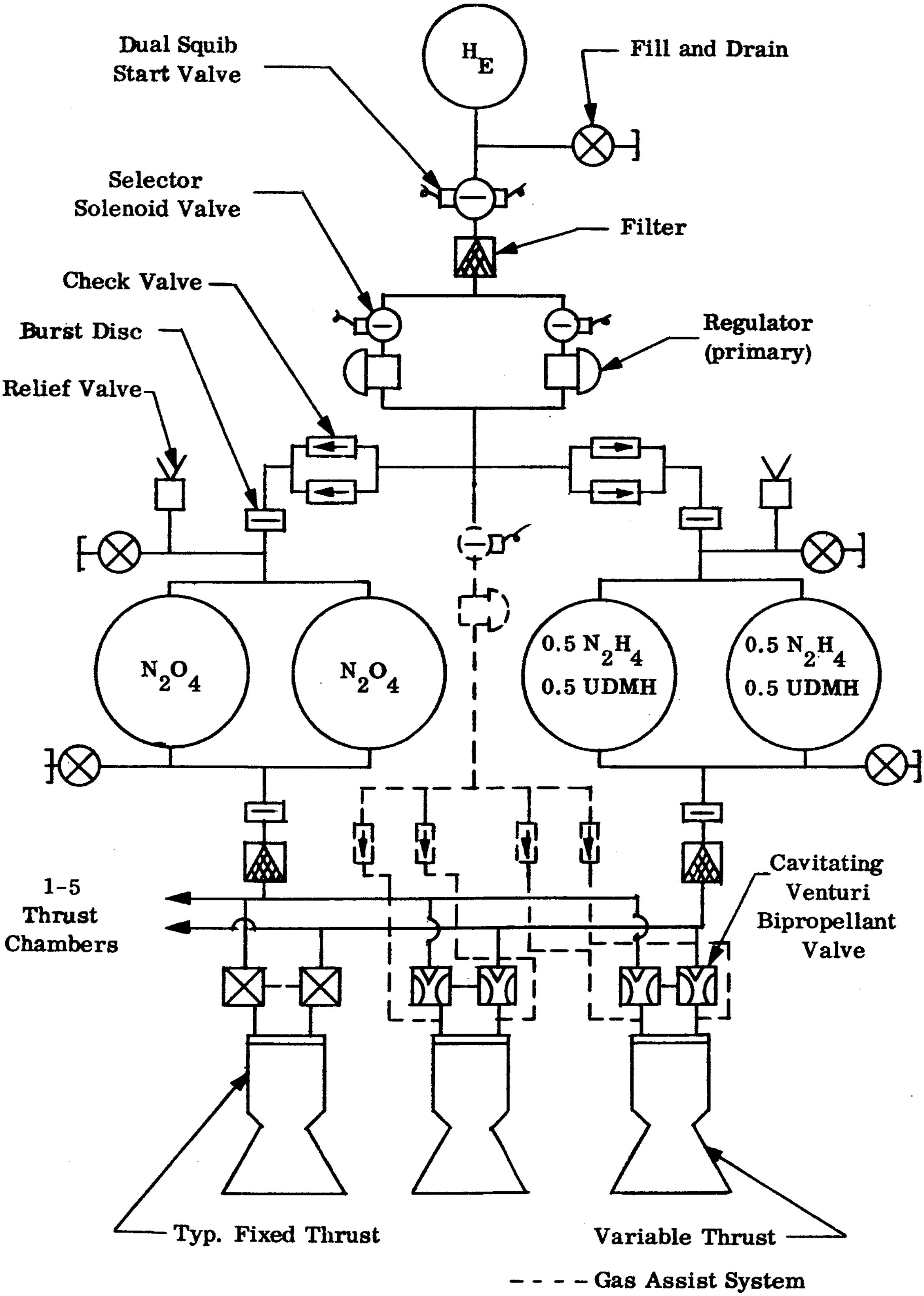


Figure 35. N_2O_4/50 UDMH + 0.5 N_2H_4 Schematic Diagram

aluminum tanks with bladders for positive propellant acquisition which are pressurized by regulated helium from a 3000 psia (max) source stored in a spherical titanium bottle. Burst discs are used upstream of the propellant filters to eliminate potential long term propellant leakage through the bipropellant thrust chamber valves and to contain the propellant within the tank to minimize freezing in the potentially exposed feed lines. Burst discs are also employed upstream of the vent and relief valves to isolate the pressurization system before launch, thereby preventing excessive entrapment of the propellants on the gas side of the bladder by condensation of the permeated propellant vapor during low temperature conditions in the lines (not thermally shorted to the propellant bulk). These burst discs in combination with the forward parallel redundancy check valves prevent propellant vapor exposure of the regulators or reactive mixing of these vapors prior to initial pressurization.

Redundant pressure regulators with selector solenoids are used together with a dual squib operated start valve to contain the high pressure helium until the system is activated. Further use of functional redundancy is not warranted without a detailed analysis of the required mission reliability and crew safety aspects. For example, the emergency shutoff valves could be used in place of the propellant side burst discs to enable cutoff if a thrust chamber valve fails to close, but it also requires actuation power, presents a potential long term leakage problem, and is a series item which must function to activate the propulsion system.

The preliminary estimate of the requirements imposed on the propulsion system is shown in Figures 36 through 39. This data is used as the initial estimate of the vehicle gross weight to define the thrust level and propellant load required by the subject mission. Figure 36 shows the lunar surface lift-off weight as a function of mission velocity increment and nonpropulsive system weight (W_{NP}) for a thrust to weight ratio of 0.5. Figure 37 shows the effects of increasing the initial thrust to weight ratio for the range of nonpropulsive weights to achieve an 8000 fps velocity increment. Figure 38 describes the thrust requirements as a function of mission Δ V for T/W's of 0.2, 0.5 and 1.0 nonpropulsive weights covering both one and two man vehicles. The dashed lines shown on these curves present a typical example for estimating the vehicle gross weight, usable propellant weight, and the effect of thrust to weight ratio on a vehicle of specified Δ V capability. Figure 39 describes the minimum firing durations to accomplish the required mission Δ V. Also shown on this figure are representative ROM estimates of the firing duration to show the effects of throttling for a constant altitude mission profile and a ballistic profile for the transportation mission with a T/W initial of 0.5; i.e., T/W = 0.5 at 100% rated thrust.

3. Propulsion Requirements

A summary of the propulsion requirements used for the N_2O_4/0.50 N_2H_4 + 0.50 UDMH system analysis are as follows:

(1) The required vehicle thrust levels are defined by the design T/W using the estimates of vehicle lunar lift-off weight in earth pounds shown in Figure 36.

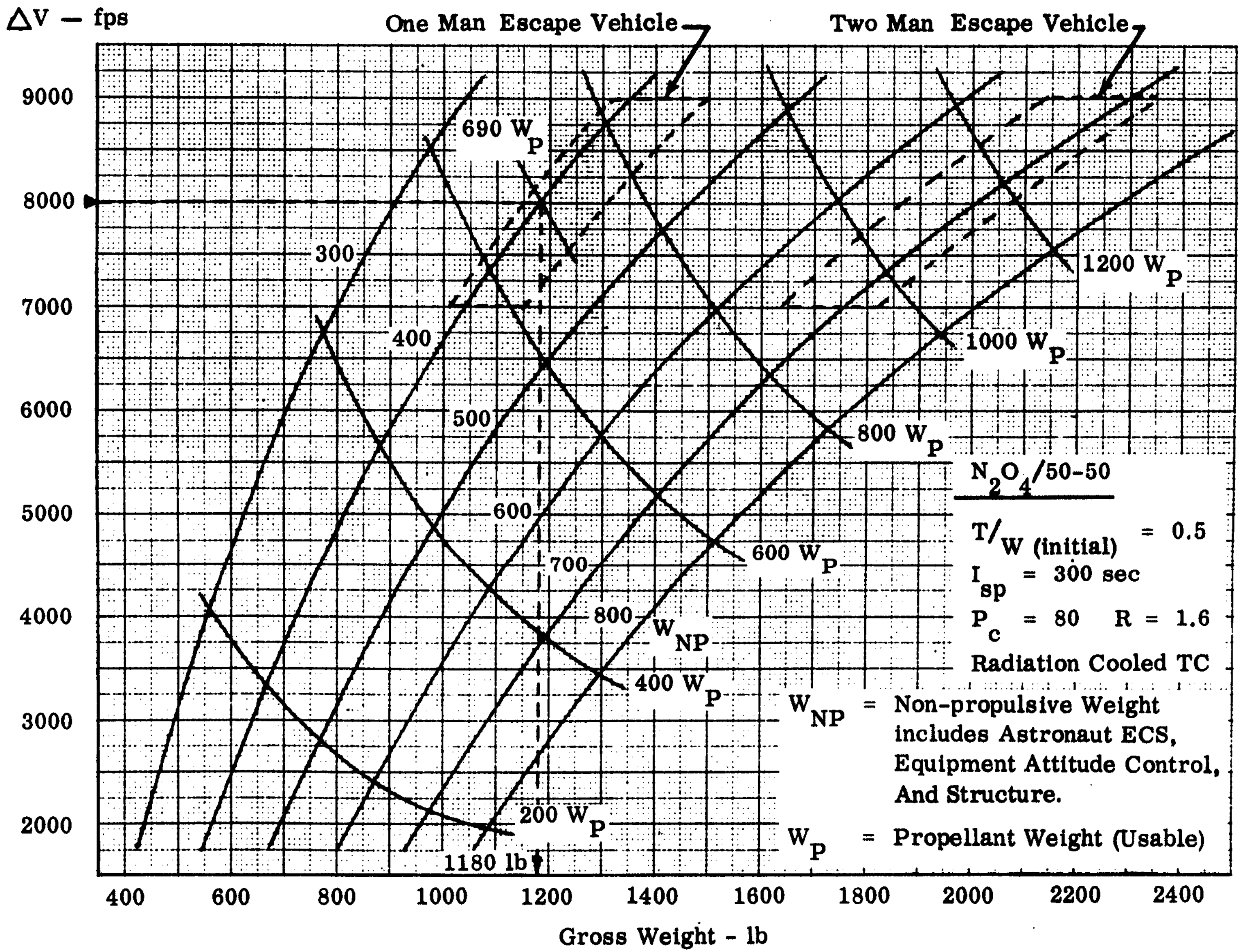


Figure 36. Gross Weight versus ΔV

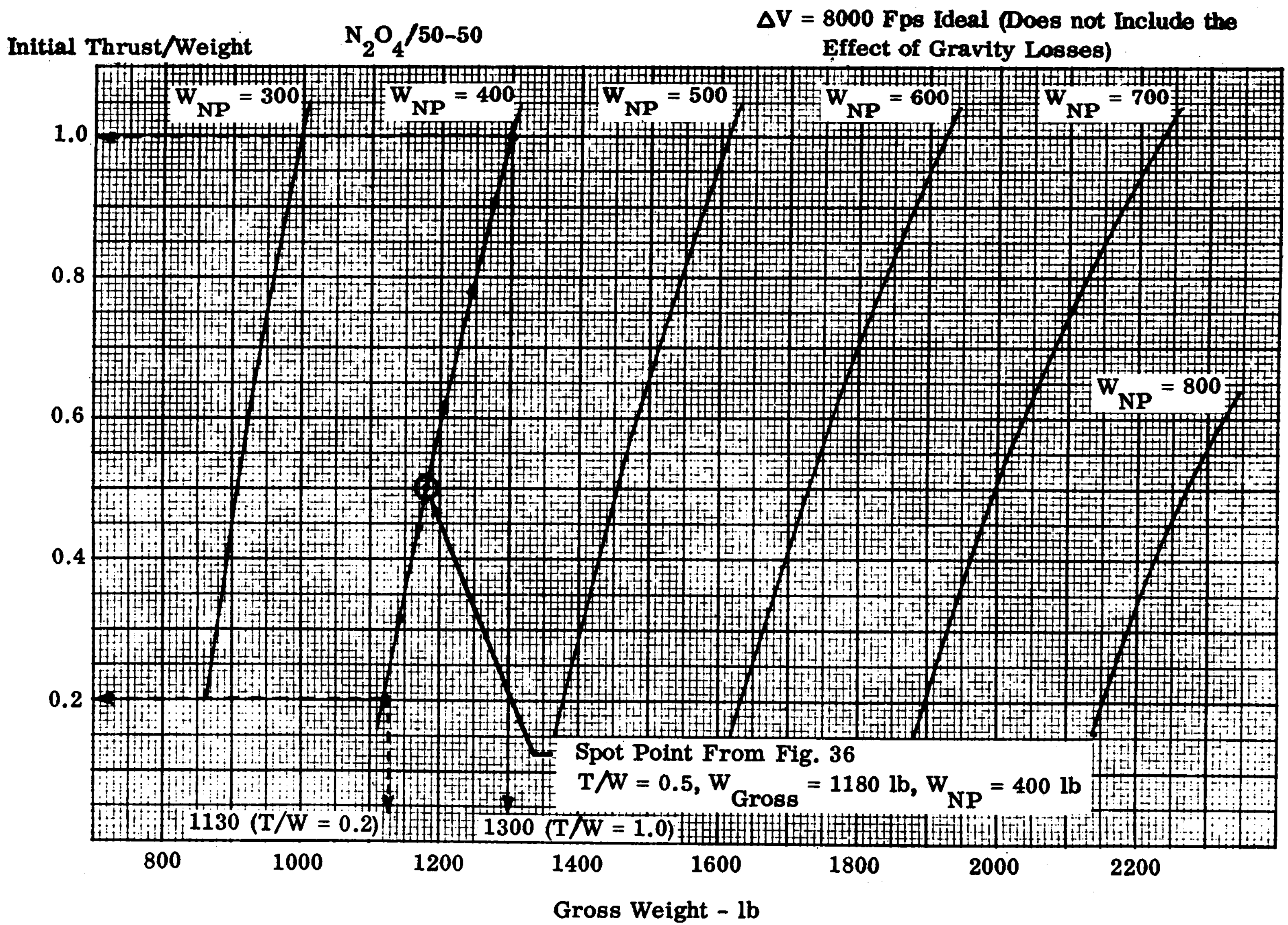


Figure 37. Gross Weight versus T/W

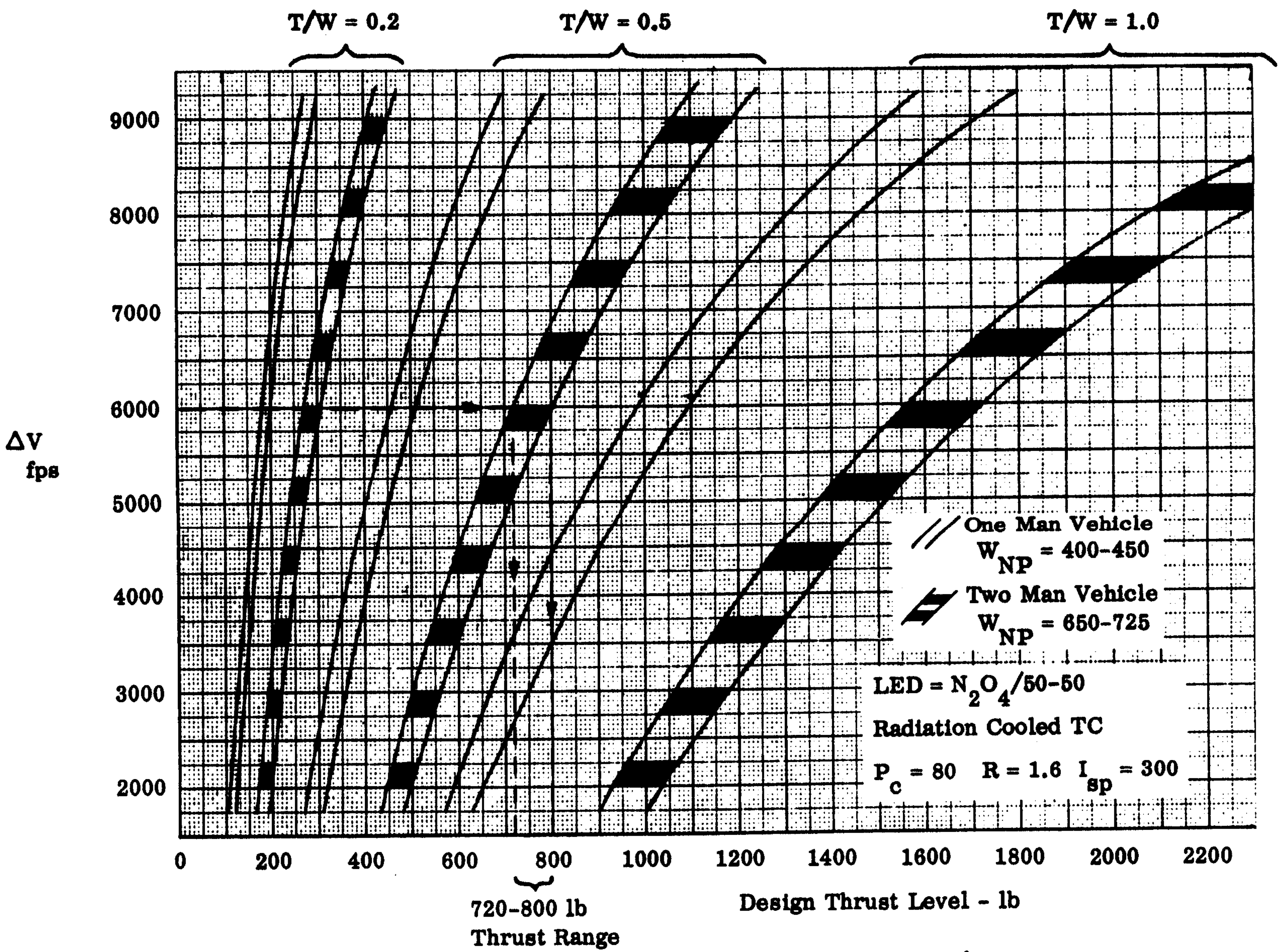


Figure 38. PPD Design Thrust Level versus Mission ΔV

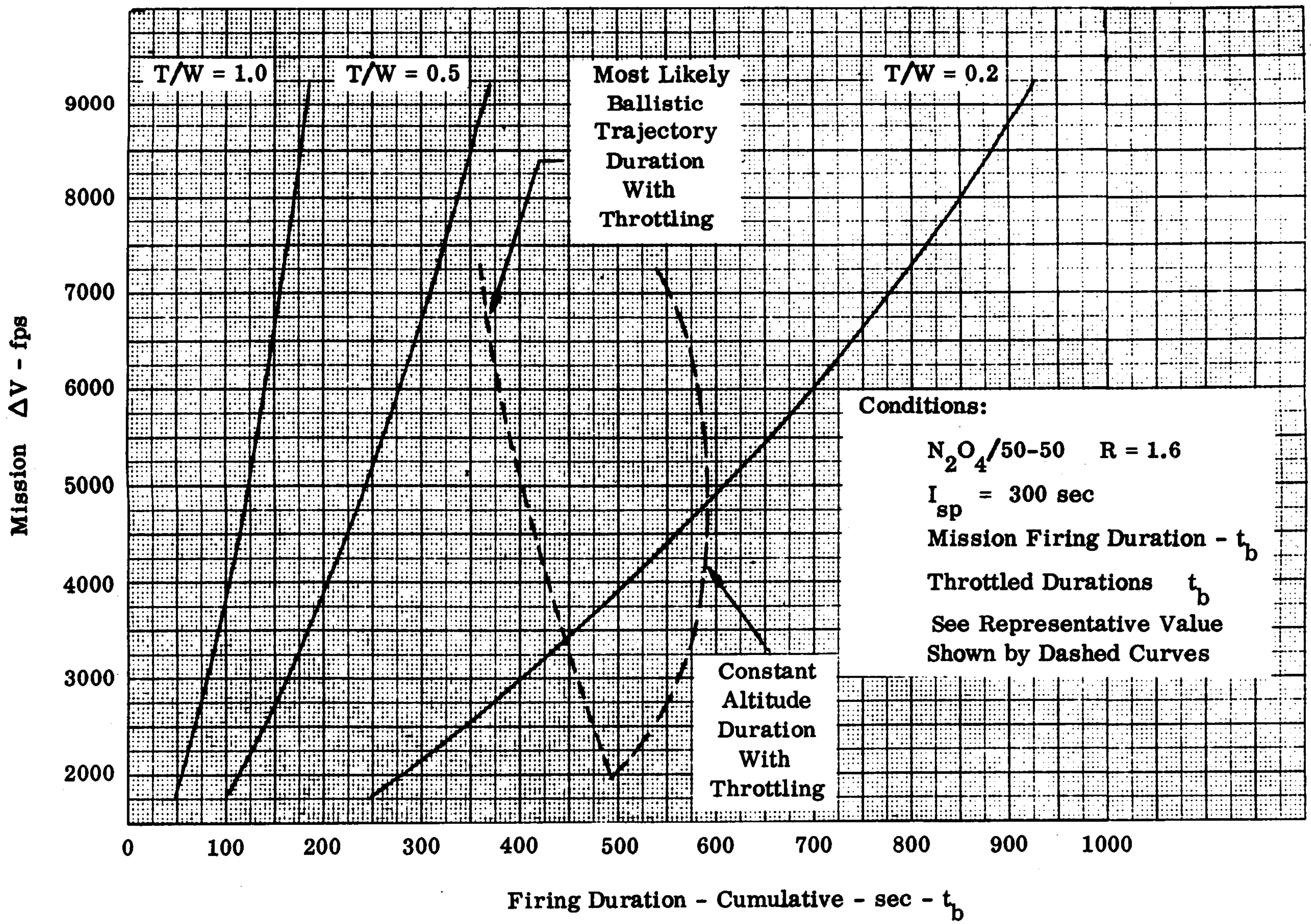


Figure 39. Minimum Cumulative Firing Duration

(2) Vehicle throttling capability of 10 to 1 is required for the transportation mission.

(3) Precise control of thrust vector magnitude and alignment is required to establish the vehicle and effects of varying T/W during simulation.

(4) Firing durations range from 150 to 1000 seconds for the escape mission, and from 50 to 2000 seconds for the combined mission capability.

(5) Investigation of crew safety aspects are limited to minimizing catastrophic failure resulting in loss of propulsion capability after lift-off from lunar surface.

These requirements are interpreted to indicate:

(1) A gas assist system is required to improve the injection characteristics in the deep throttled range (over 3 to 1) in the throttled systems while fixed orifice injectors will be used for the others. This throttling technique was chosen because it provides high performance with high reliability and does not have critical flow control problems (circa a variable area injector) at the low thrust levels required for multiple engine systems and is being developed for use on the LEM Descent System. Use of this technique requires a small increase in gas weight and the addition of some componentry; however, the increase in deep throttled performance more than offsets the additional component weight, resulting in less vehicle gross weight for most missions.

(2) A radiation cooled thrust chamber offers distinct advantages for this application in the areas of thrust chamber weight and by avoiding problems in thrust vector magnitude and alignment variations with time, when compared with an ablative chamber with potential throat erosion during the long firing durations required. The radiation cooled chamber is less rugged from an impact damage standpoint, but offers the capability of reuse and has been extensively developed over the required thrust range. Injector characteristics are essentially the same for either type chamber and the performance will be similar; however, the ablative will suffer some loss if throat erosion occurs. The radiation chamber will suffer none of the problems with outgassing at shutdown for the transportation mission. Both types can employ radiation cooled nozzle extensions to minimize weight, but require consideration during installation to avoid radiation interchange overheating problems. An all ablative chamber is recommended for buried installations. The ablative chamber offers a smaller envelope by operating at higher chamber pressures than permitted by the cooling mechanism for the radiation chamber. Both types use a fuel rich barrier to reduce wall heat transfer rates.

(3) Throttled operation will use variable area bipropellant valves designed to cavitate below approximately 80% thrust to minimize propellant residuals (constant mixture ratio) and to insure combustion stability with the gas-assisted injection technique. The fixed thrust system uses start/stop valves in place of the venturis with similar characteristics at rated flow, which results in a common thrust chamber design for both fixed and throttled systems with the exception of the gas inlet.

C. OPERATING LEVEL OPTIMIZATION

An analysis was conducted to determine the operating pressure levels which minimize the lunar lift-off weight for the candidate configurations. This was used to determine the chamber pressure and area ratio best suited for the specified missions, and to define the penalties incurred from operation outside this optimum envelope. A single chamber pressure of 80 psia was chosen for all thrust levels (based on this analysis and the cooling technology limitations) which provided the basis for elimination of rated chamber pressure as an independent variable in the final system parameters.

The approach employed for the chamber pressure optimization study was to divide the vehicle lift-off weight into three groups, defined by the parameter which primarily establishes the weight of the components in that group. One group consists of the propellant feed system major components — propellant, propellant tanks, bladder, pressurant, and its tank. Another consists of the components which are a function of the thrust level and number of thrust chambers; which includes the thrust chambers, their valves, the regulators, the feed lines and fittings, and the chamber mounting equipment. The balance of the vehicle components is in the last group, including the payload, basic vehicle structure, the astronaut and his equipment, the guidance and electrical equipment, and the fixed weight propulsion system componentry — relief valves, fill and vent valves, pressurant start valves, etc.

The analytical technique employed consists of solving for the lunar lift-off weight W_V. The methodology can best be illustrated by example - Case No. 3.

Conditions

ΔV = 6000 fps

W_{NP} = 450 lb (corresponds to one-man vehicle)

F = 500 lb single chamber T/W = 0.5

W_P propellant burned - 475 lb

(These starting conditions were obtained using Figure 36 to relate ΔV to W_{VO} for desired T/W).

$W_O \quad = W_{NX}/(1 - \delta_V/C)$

$W_{NX} \quad = W_{NP} + W_{XC} + W_{TC}$

$W_{NP} \quad =$ nonpropulsive weight = 450 lb

$W_{TC} \quad =$ thrust chamber weight = (No TC) (W/TC) (See Figure 40)

$W_{XC} \quad = W_{\text{press. components}} + W_{\text{feed system components}}$

$\qquad + W_{\text{TC valves}} + W_{\text{instrumentation and control}}$

$\qquad + W_{\text{thrust mount}} = 31.5$ (See Table I)

$W_{NX} \quad = 450 + 31.5 + W_{TC}$

$\delta_V \quad = 1 - \exp(-\Delta V/gI_{s\ theo}\ N_I)$

$g \quad = 32.174\ \text{ft/sec}^2$

$I_{s\ theo} \quad =$ theoretical Bray I_{sp} at P_c and ϵ from Figure 43

$N_I \quad = I_{sp}$ efficiency at F/TC level from Figure 44

C from Figure 45 for $W_{PE} = 475$ at $P_T = P_c + 60$ psia

The parametric weight estimate of the thrust chamber is shown in Figures 40 and 41 for thrust levels of 500 and 100 lb, respectively, for a range of chamber pressures and area ratios from 20 to 120. This data resulted from thrust chamber designs for 20, 40, 80, and 120 area ratio at 20, 40, 80, and 120 psia chamber pressures for both thrust levels. These designs employed columbium thrust chamber material to a point in the nozzle where the local unit weight for columbium equaled the local unit weight for titanium. The balance of the nozzle to the exit was titanium, employing a bolt-on flange type connection. Fixed orifice injector design was utilized for both throttled and fixed thrust designs with the gas assist technique employed for the deep throttling requirements. This resulted in similar designs for both throttled and fixed thrust chambers with equivalent weights. The weight estimates are based on chamber geometries defined by the BAC Combustion Devices Group, with chamber wall thicknesses dictated by lateral conductivity requirements (to prevent hot spots); and the stiffness requirements, or the usable hot strength stress levels, based on the wall temperature data presented in Figure 42. Figure 42 described the average wall temperature at each station throughout the nozzle with the throat temperature used for the wall of the combustion chamber (sections upstream of throat) for design purposes. This figure is an extrapolation over the design range of the test data from the current BAC 100-lb Thrust Chamber Development Program.

The weights of the system components were estimated by similarity with equivalent systems as shown in Table I. This table lists the components described in the representative system schematic for a fixed thrust system, shows the number required, and the parameter which establishes its weight. The estimates given are for Case No. 3, used as an example in the discussions presented later in this section.

TABLE I

PROPULSION SYSTEM COMPONENT WEIGHT ESTIMATE

Nomenclature	No. Required	Weight	Influence Parameter: Propellant Expelled	No. TC	Thrust
PRESSURIZATION SYSTEM					
Gas Bottle	-	-	x		
Fill & Drain Valve	1	0.3			
Start Valve (dual squib)	1	1.0			
Filter	1	0.4			x
Regulator Selector Valve	2	3.6			x
Regulator	2	2.0			x
Check Valves (parallel set)	2	2.6			
Isolation Burst Discs	2	0.4			x
PROPELLANT FEED SYSTEM					
Oxidizer Tank (with bladder)	-	-	x		
Fuel Tank (with bladder)	-	-	x		
Vent Valve	2	0.6			
Relief Valve	2	2.6			
Fill & Drain Valve	2	0.8			
Burst Discs	2	0.6			x
Filter	2	0.8			x
Lines and Fittings	-	7.5	x	x	x
THRUST CHAMBER ASSEMBLY					
Thrust Chamber	-	-	-	x	x
TC valves (bipropellant)	1	1.8		x	x
Mounting Equipment	-	4.5		x	x
Gas Assist System — Not included in optimization:					
Start Valve		1.0			x
Regulator		1.6			x
Check Valves		0.6 per set		x	x
Lines					
Instrumentation		2.0			
Total (Minor comp) (Case No. 3)		31.5			

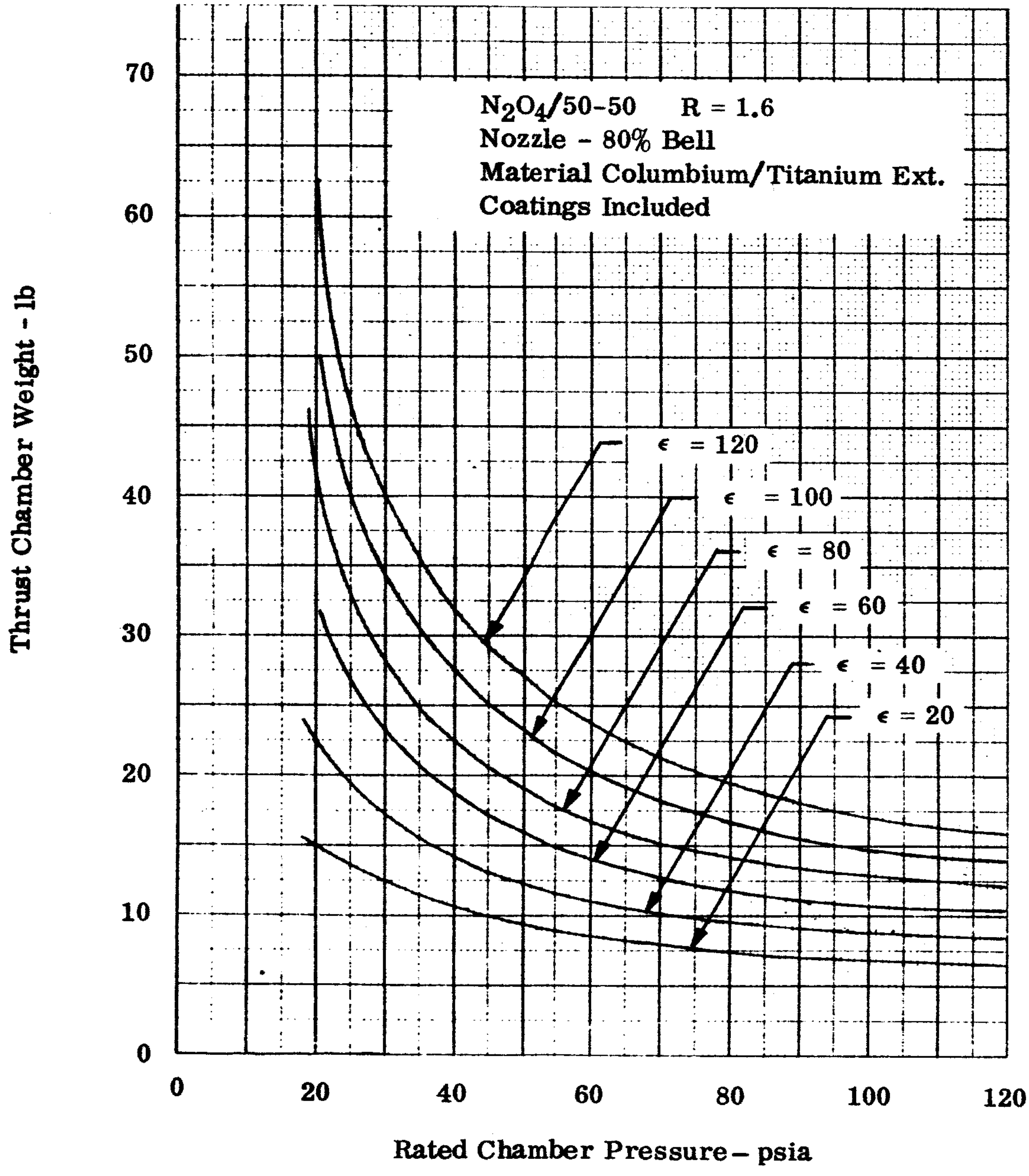


Figure 40. 500 lb Radiation Cooled Thrust Chamber Weight versus Chamber Pressure and Area Ratio

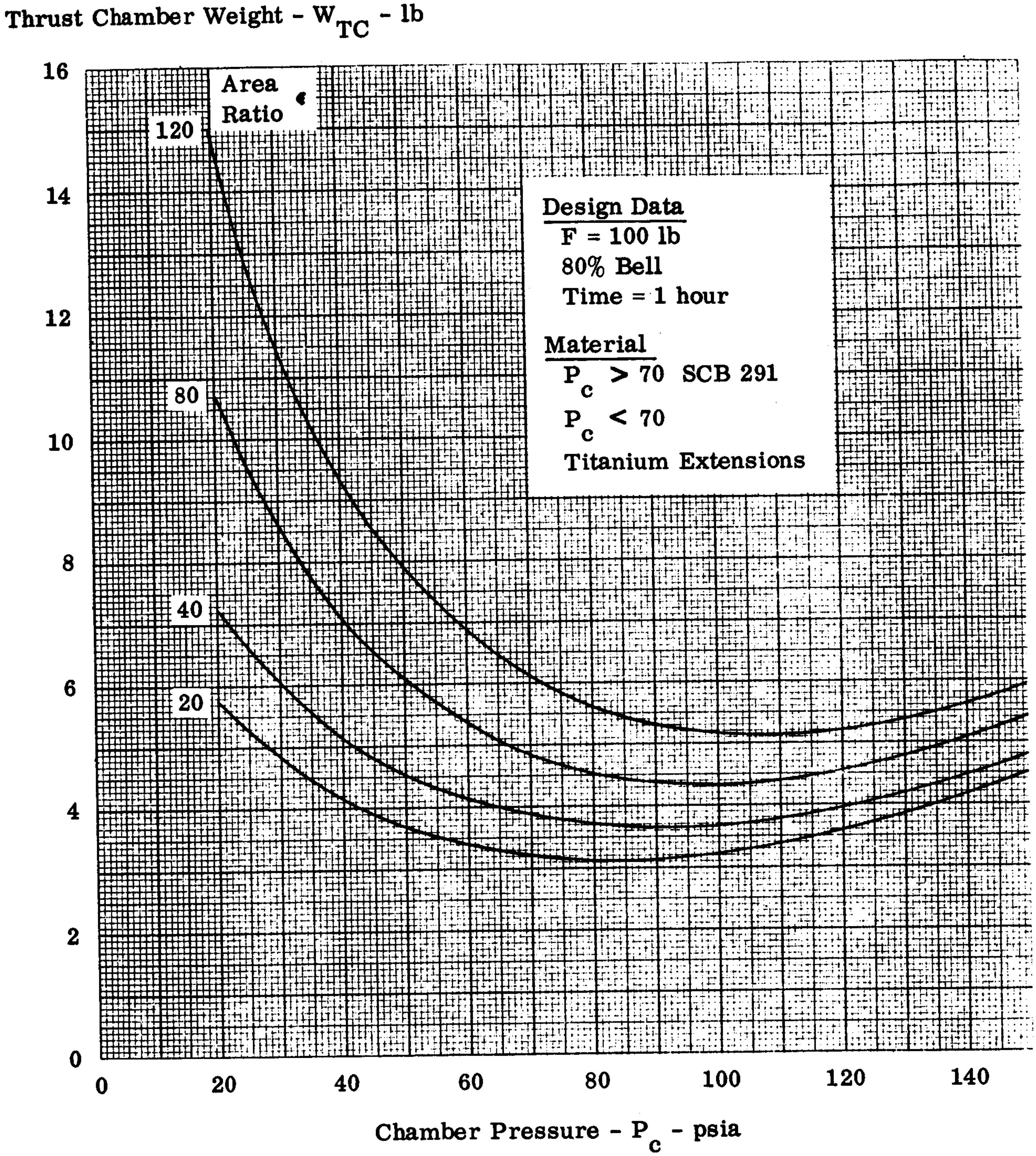


Figure 41. 100 Pound Radiation Cooled Thrust Chamber Assembly Weight versus Chamber Pressure and Area Ratio

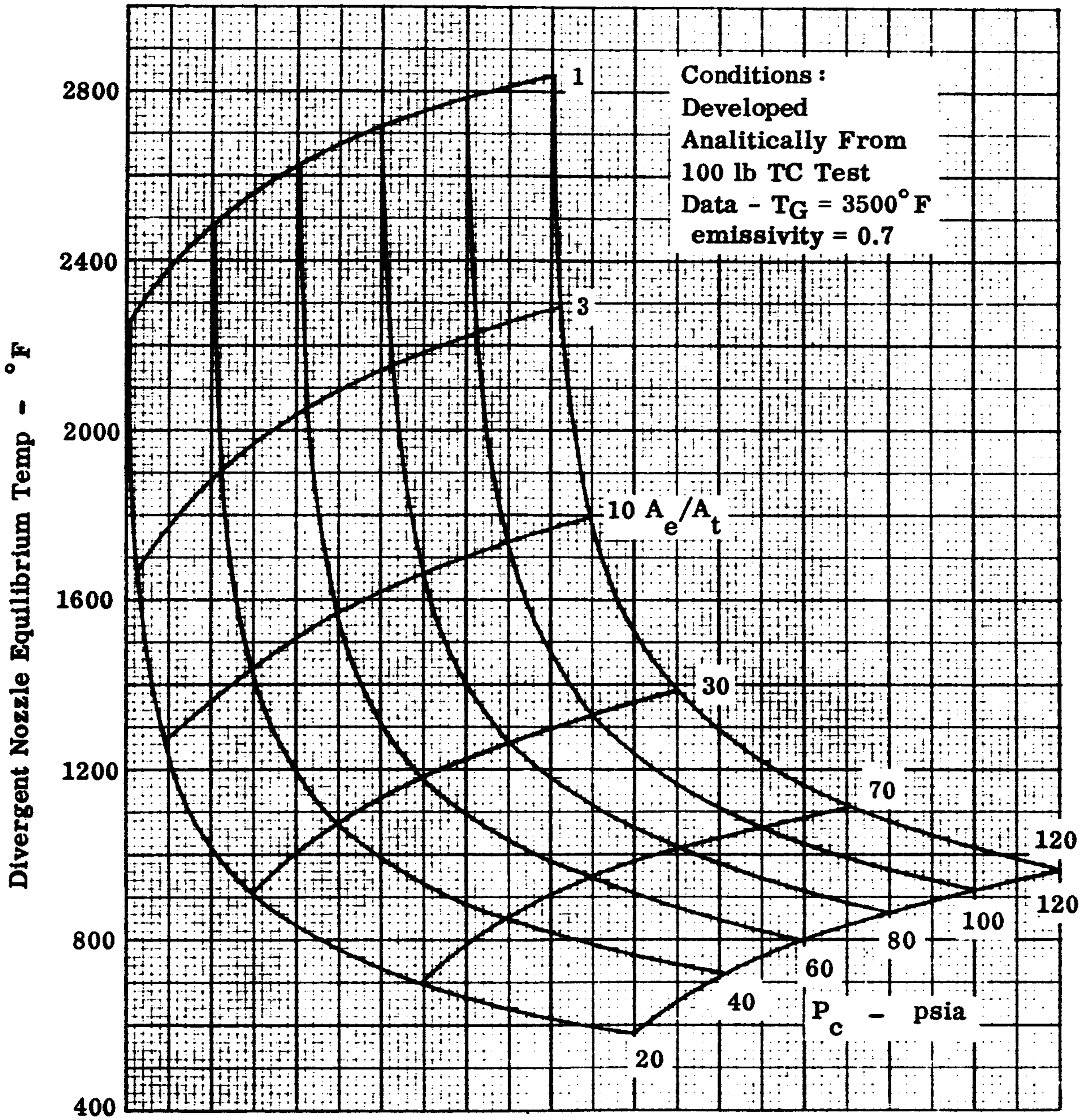


Figure 42. Divergent Nozzle Equilibrium Temperature versus P_c and A_e/A_t

Figures 43 and 44 are used to define the delivered specific impulse in terms of the best estimate of the scale effects on performance for the range of interest for this study. Figure 43 presents the theoretical Bray data for a thrust level of 3750 lb using a Model 8173 Maneuvering Satellite nozzle contour. The efficiencies shown in Figure 44 were obtained by dividing the nominal delivered performance obtained from test data into the Bray performance of Figure 43, and smoothing of the curve using the best estimates from the Combustion Devices Group for 100, 500, 1000, and 2500 lb thrust levels at the design conditions for this application (R = 1.6, radiation cooled TC, 8173 type nozzle, etc.). This efficiency is for rated thrust only with an efficiency as a function of throttle depth defined as shown on Figure 50.

The weight of the propellant feed system major components has been defined as a function of the propellant weight required to achieve the design velocity increment. The ratio of propellant burned to propellant loaded, plus tanks, bladder, and pressurant is defined as "C" for the purposes of this study, as shown in Figure 45. Equal volume spherical propellant tanks of 6061-T6 aluminum with 0.009 in. thick teflon bladders are used for the propellants, and sized to provide 2.0% ullage with 100% propellant load at 580°R. Of this load, 98% is available for propulsion. A 0.005 in. scratch allowance on wall thickness sized for stress,using a factor of safety of 2.00 on an ultimate tensile strength of 44,000 psi, with a minimum wall thickness of 0.020 including scratch allowance, was used for propellant tank weight estimates.

Regulated helium stored in a single spherical tank is used to pressurize the propellants. The pressurant requirements were sized with 30% reserves using a k of 1.2 based on propellant tank volume from a source of 3000 psia max to 400 psia min at 580 and 480°R, respectively. The gas assist requirements and the pressurant for the attitude control system is included in the 30% reserve for optimization purposes. The pressurant requirements are based on the required propellant pressure (P_T) in the tank, and the tank shell weights are based on this pressure plus 60 psia for regulator and relief valve tolerance stackup allowance.

Representative vehicle configurations were chosen for both the transportation and escape missions to cover the range of vehicle parameters which influence the optimum chamber pressure. The vehicle parameters and the results applicable to the escape mission are shown in Table II. Two cases at each of the specified thrust to weight ratios were investigated using 1 to 4 main thrust chambers and representative vehicle weight parameters for one and two man missions. The results are shown in Figures 46 and 47. These curves describe the ratio of the lunar launch weight required by the chamber pressure and area ratio in question to the minimum launch weight observed, expressed in percent, for the mission parameters investigated. Thus, the optimum chamber pressure is determined and the launch weight increase due to operation at other pressures or area ratios is defined. Figure 48 is a plot of the 80 psia chamber pressure data against area ratio as an aid to the evaluation of envelope limitations on vehicle performance.

TABLE II

ESCAPE MISSION P_c OPTIMIZATION

Case No.	T/W	No. TC	F/TC	I_{sp} Efficiency	W_{NP}	Weight Components	Type Vehicle	ΔV	Results $(W_o)_{min}$	$(P_c)_{opt}$	ϵ_{opt}
1.	0.2	2	100	0.8985	480	20.0	one man	6000	1007	83	120
2.	0.2	4	100	0.8985	720	30.2	two man	8000	1928	83	120
3.	0.5	1	500	0.9193	450	31.5	one man	6000	980	83	80
4.	0.5	3	225	0.9095	450	36.0	one man	8000	1296	78	60
5.	1.0	1	2000	0.9352	650	59.2	two man	8000	1864	80	40
6.	1.0	4	500	0.9193	625	80.2	two man	8000	1924	90	60

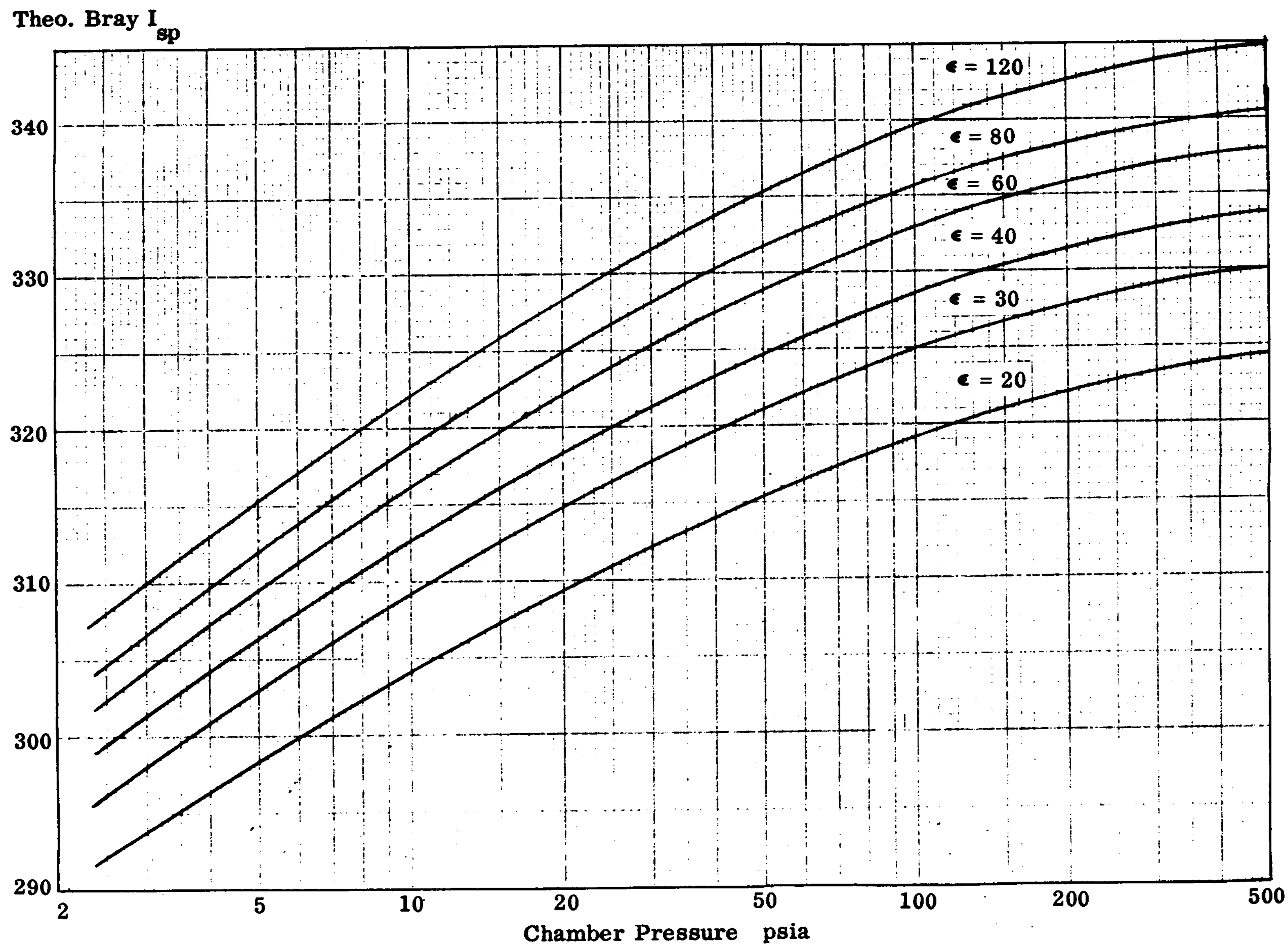


Figure 43. Theo. Bray I_{sp} versus P_c $N_2O_4/0.50\ N_2H_4 + 0.50$ UDMH R = 1.6

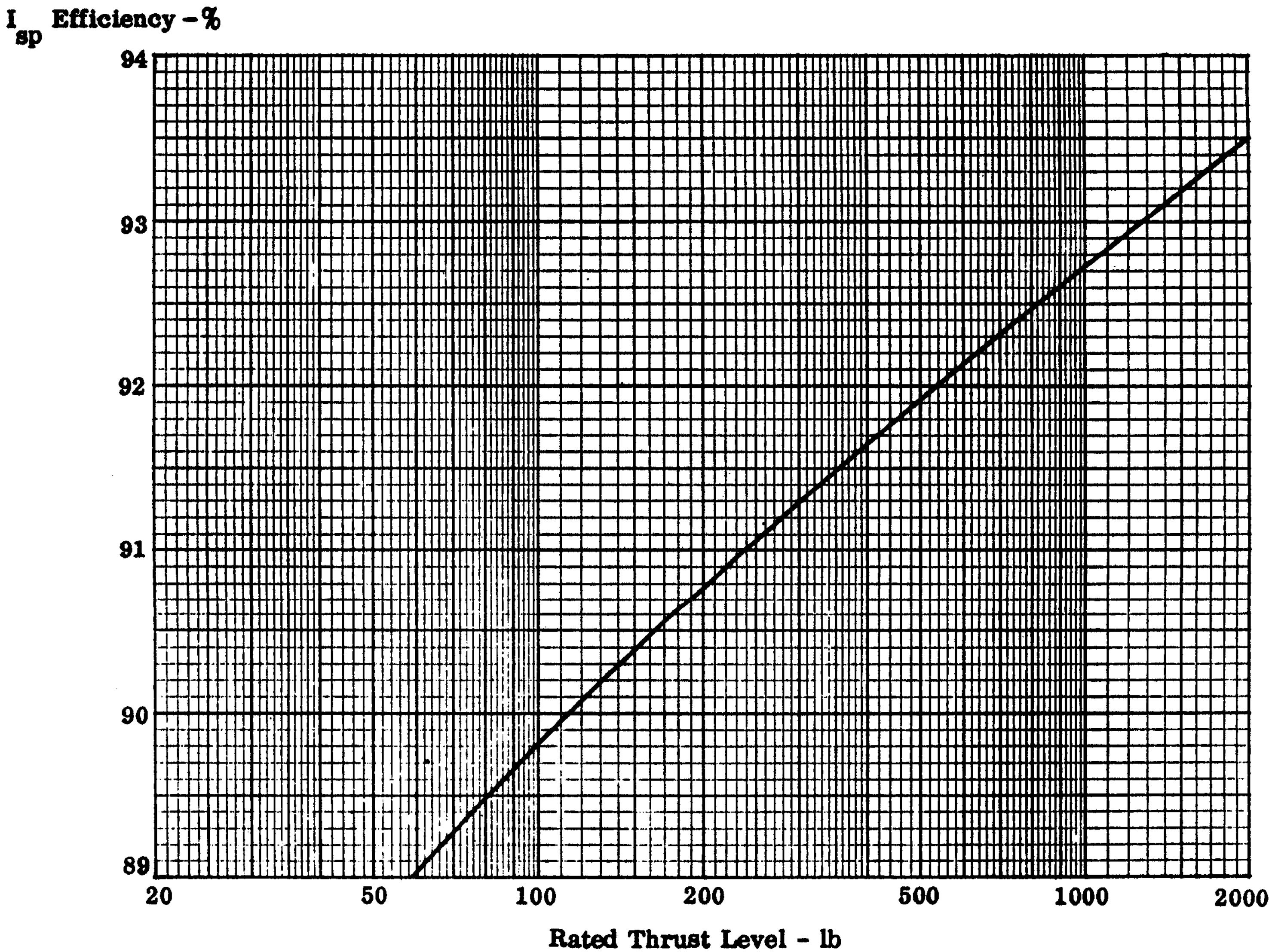


Figure 44. Thrust Level Performance in Percent of Theo. Bray I_{sp}

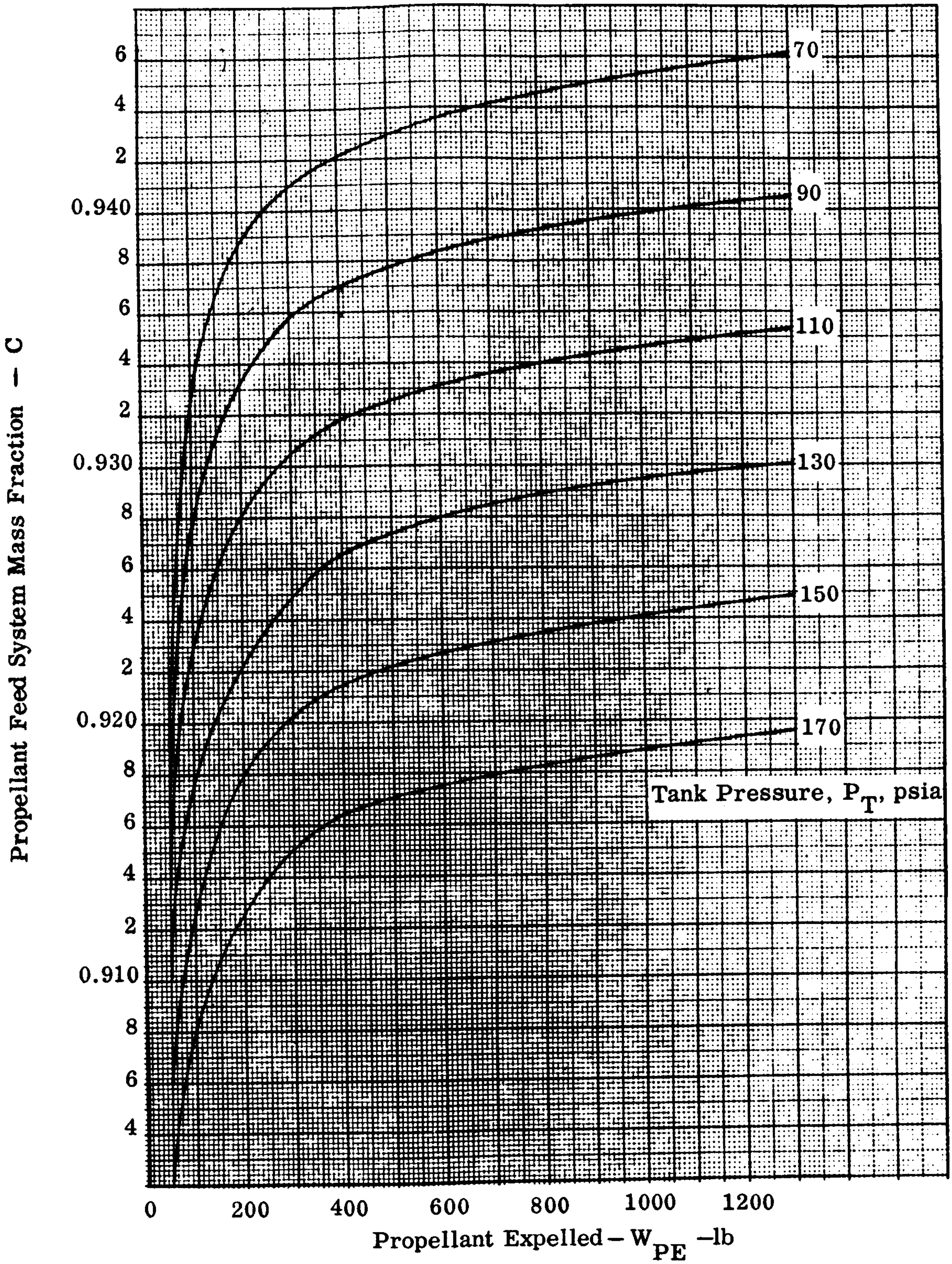


Figure 45. Propellant Mass Fraction

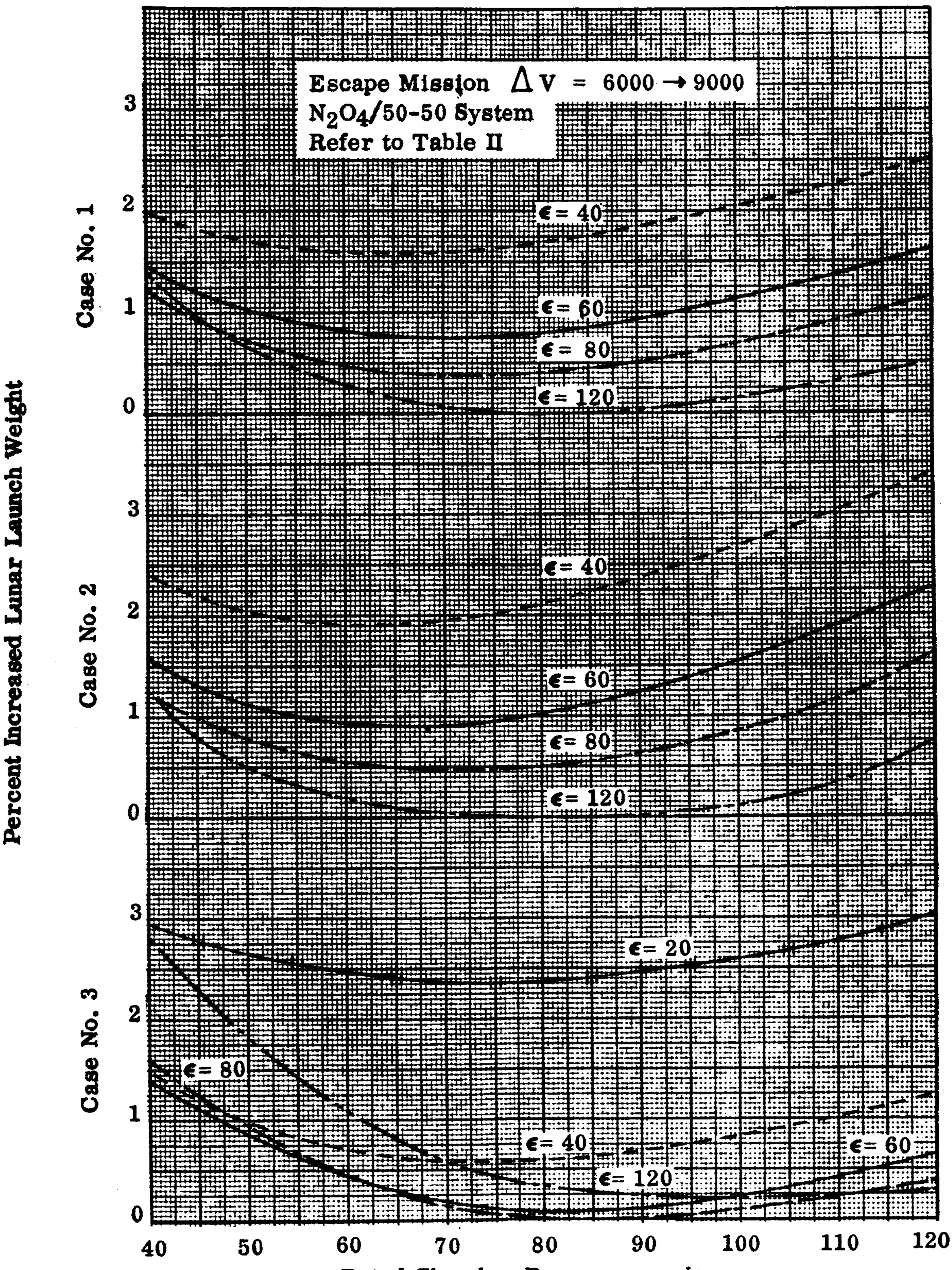


Figure 46. Chamber Pressure Optimization

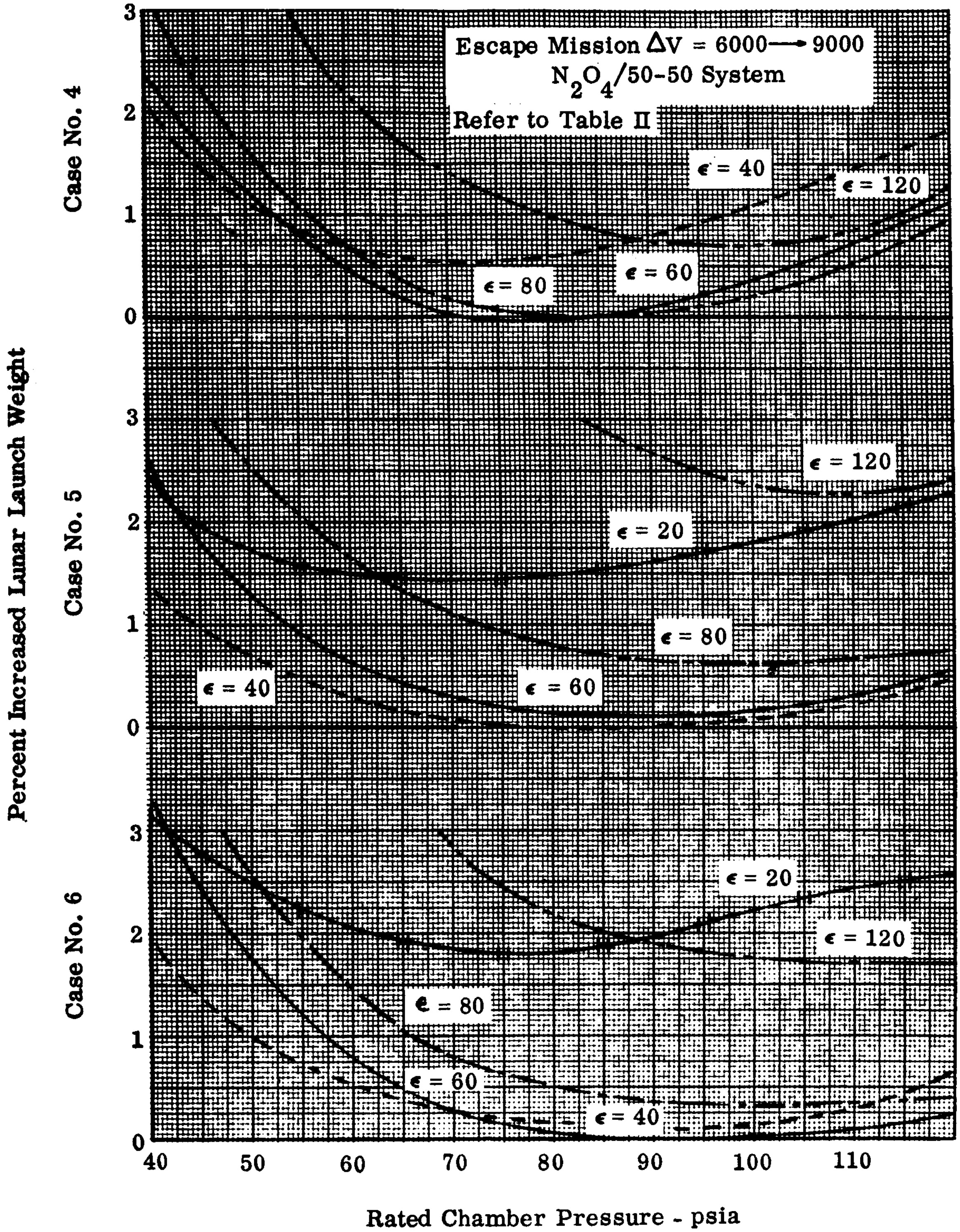


Figure 47. Chamber Pressure Optimization

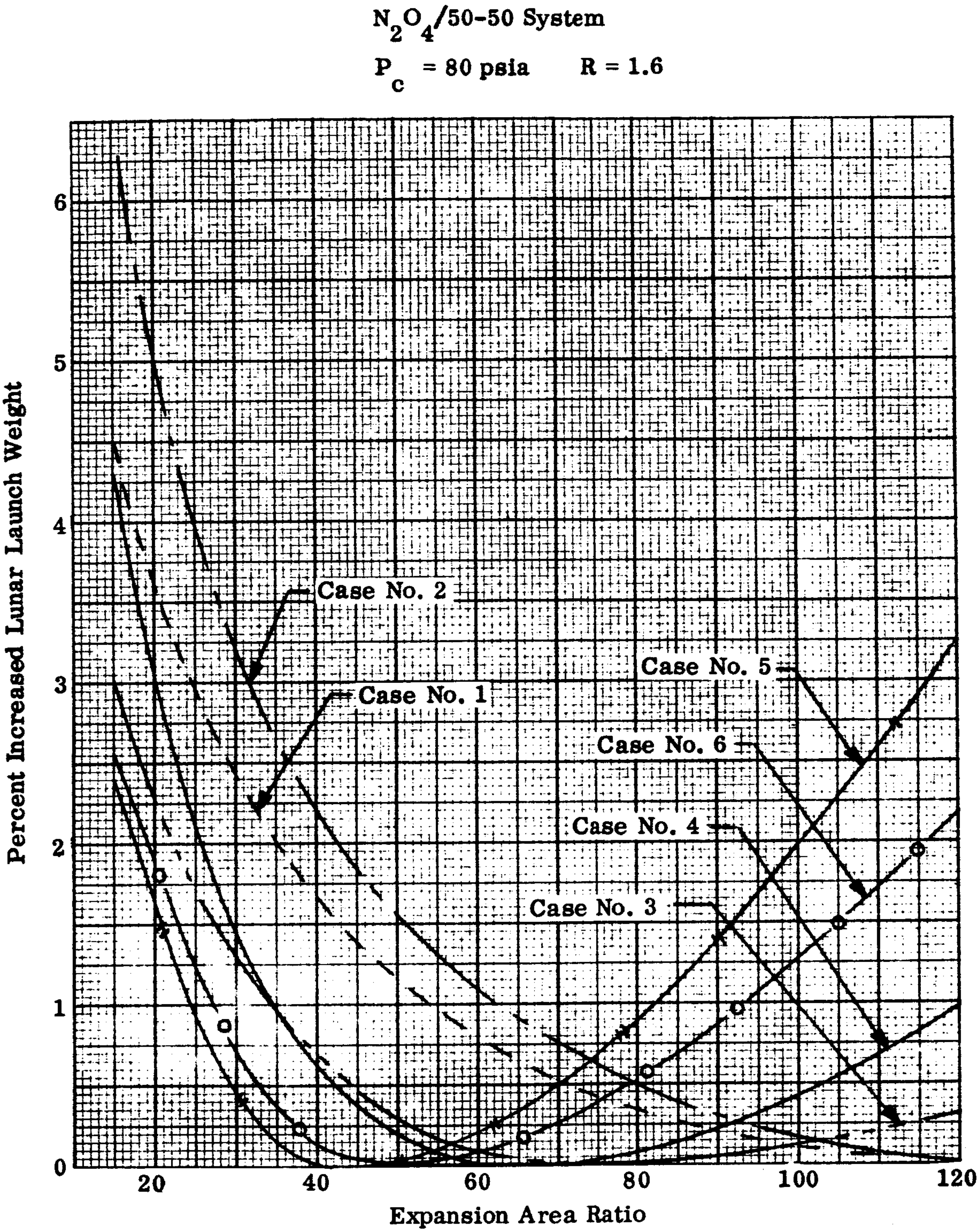


Figure 48. Expansion Area Ratio Effects

The results of the optimization for the transportation mission are shown in Figure 49 and Table III for velocity increments of 2000 through 8000 fps at a thrust to weight ratio of 0.5. This data shows that an area ratio from 40 to 100 results in under a 1% launch weight variation at the 80 P_c optimum pressure for the transportation mission. This indicates that the thrust chamber envelope constraints and vehicle structural weight requirements will define the optimum area ratio for this mission.

D. THRUST CHAMBER PARAMETRICS

The delivered performance, weight, and envelopes were determined as a function of rated thrust and expansion area ratio (A_e/A_t) for ablative and radiation cooled thrust chambers operating with N_2O_4 and hydrazine blend (0.50 N_2H_4 + 0.50 UDMH) propellants at a mixture ratio of 1.6 and at (or near) the optimum chamber pressure for the anticipated Personnel Propulsion Devices application.

It was necessary to use a common design philosophy over the complete thrust range to insure the significance of the mission simulation results when the relative mission differences are used for system selection. Validity of the data over the complete thrust spectrum was obtained by using point designs at the optimum conditions with a "best engineering estimate" to interconnect the points and complete the curves.

1. Performance

The variation of thrust chamber performance with thrust level and expansion area ratio is given in Figure 50. The performance as measured by delivered specific impulse is given for rated thrust levels from 100 to 2600 lb and for area ratios of 20 through 120. The rated thrust is defined as the nominal thrust for a fixed thrust application, or as 100% of maximum nominal thrust for an application requiring throttling. The basic I_{sp} curves are for the radiation cooled chamber (P_c = 80 psia); however, the ablative chamber (P_c = 120 psia) was found to exhibit a 2.0 second or 0.6% increase over the complete range of Figure 50 indicated by the 100% rated thrust point on the throttling effects curve shown in Insert A. The decrease in specific impulse with throttling is shown in the throttled effects insert for both the ablative and radiation cooled chambers as a ratio of the delivered performance at the required thrust level (expressed as a percentage of rated) to the rated thrust performance. The throttled specific impulse is the product of the rated specific impulse and this ratio.

This performance data is based on the specific impulse as defined by the Bray performance prediction technique with the ratio of delivered to theoretical I_{sp} determined by extrapolation of BAC test data. This approach was used for both the rated and throttled performance to accurately describe the scale effects in terms of real hardware and to estimate the combustion efficiency losses and the low chamber pressure effects ("Bray losses") for the throttled systems. The throttled performance is based on the results of the current BAC throttling tests at the 100-pound thrust level.

TABLE III

TRANSPORTATION MISSION P_c OPTIMIZATION

Case No.	ΔV	No. TC	$F/_{TC}$	I_{sp} Efficiency	W_{NP}	Weight Components	Type Vehicle	$T/_W$	Results (W_o) min	(P_c) opt	ϵ opt
7	2000	3	100	0.8985	450	26.0	one man	0.5	613	95	60
8	4000	4	100	0.8985	450	30.2	one man	0.5	786	90	80
3	6000	1	500	0.9193	450	31.5	one man	0.5	980	83	80
4	8000	3	225	0.9095	450	36.0	one man	0.5	1296	78	60

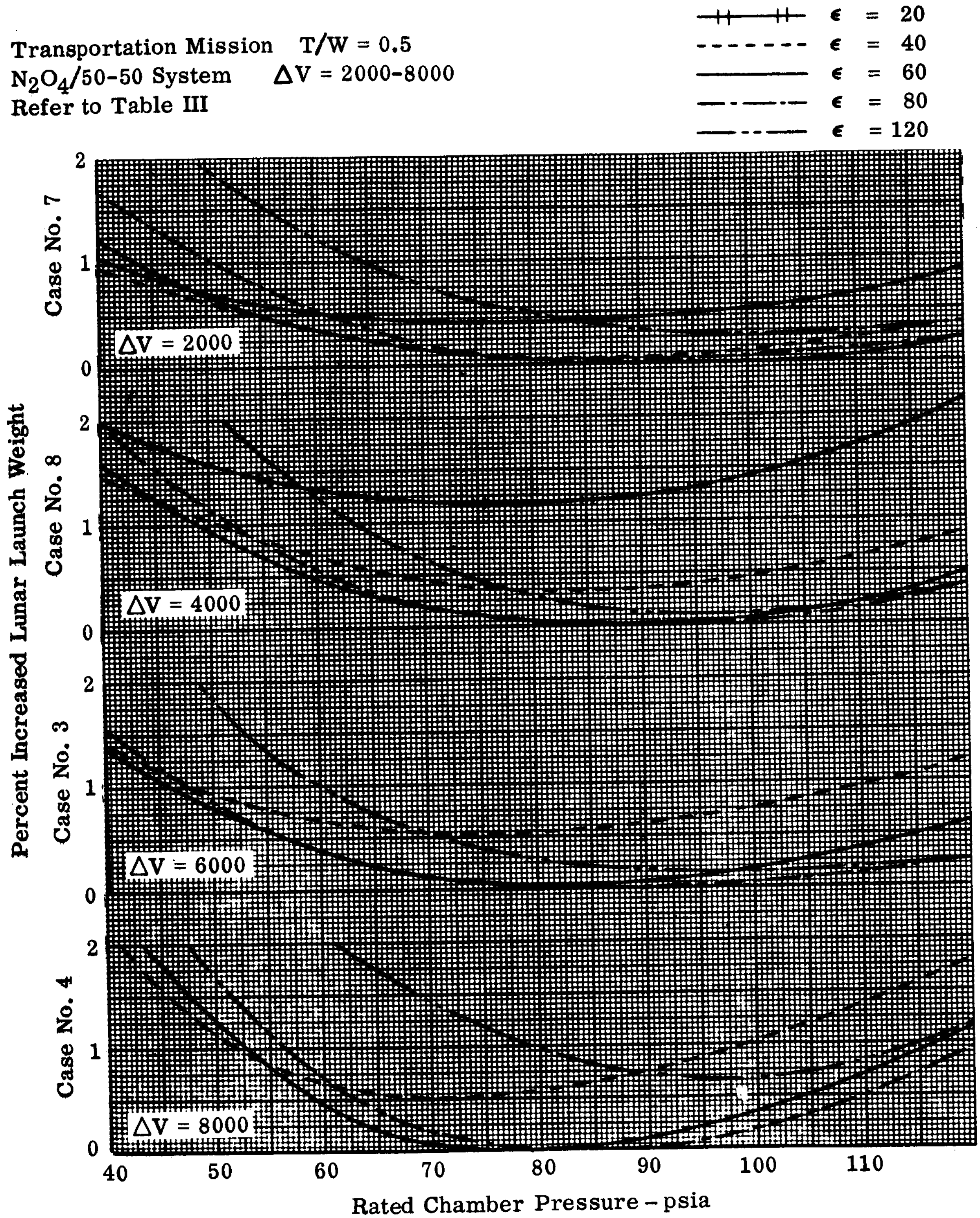


Figure 49. Chamber Pressure Optimization

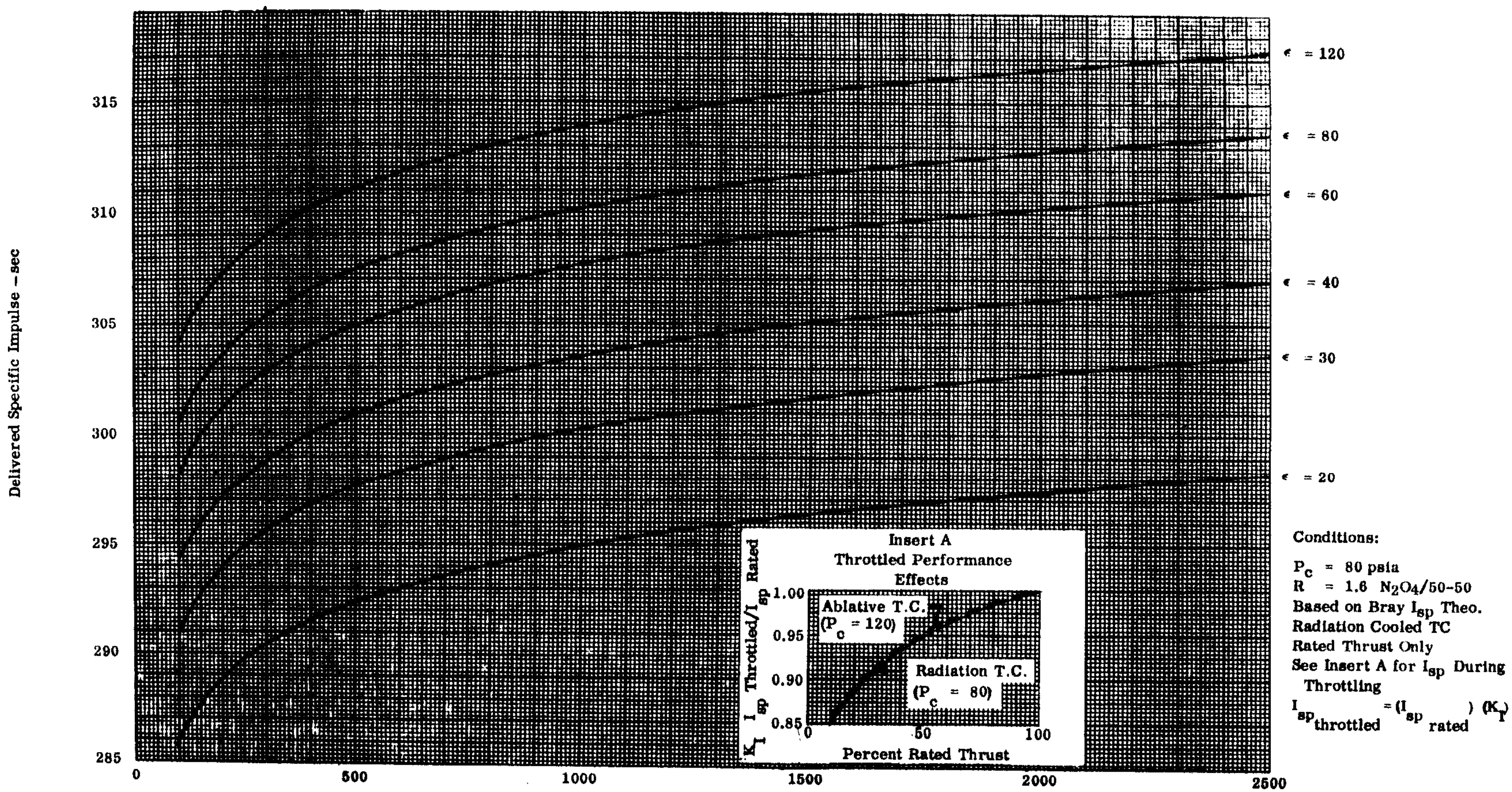


Figure 50. Delivered Specific Impulse versus Thrust Level and Area Ratio

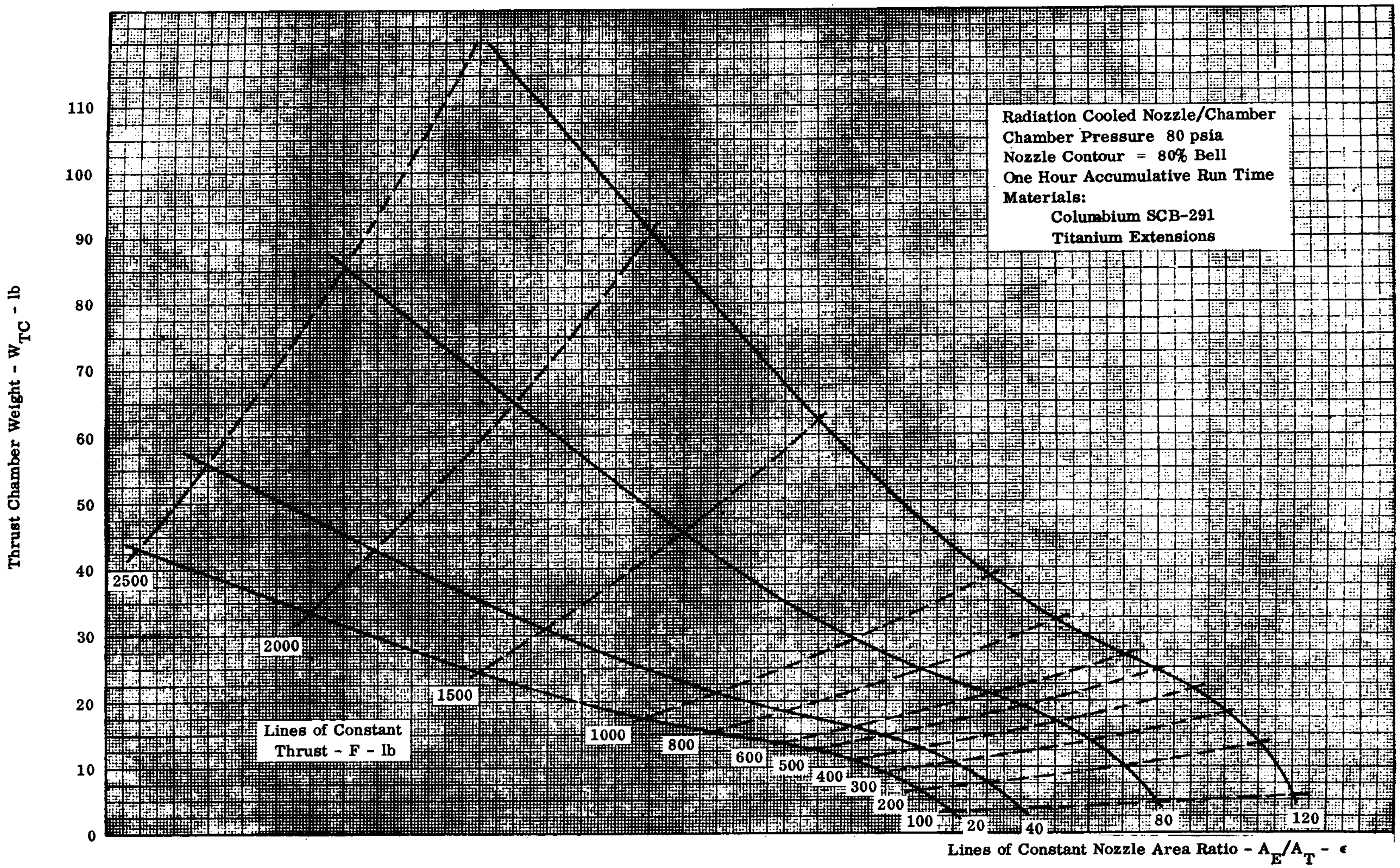


Figure 51. Thrust Chamber Weight versus Nozzle Area Ratio and Thrust Level

2. Radiation Cooled Thrust Chamber Weight and Envelope

Figure 51 describes the weight of a radiation cooled thrust chamber for the thrust levels and area ratios, and for the design conditions used for the performance data of Figure 50. This is a complete carpet plot to facilitate interpolation anywhere within the required parametric range. These weight estimates are based on point designs for this application using columbium combustion chambers, injectors, and nozzle sections; a titanium extension is used for the balance of the nozzle. The joint location for the transition from the high operating temperature material (columbium) to the lower density titanium was selected to provide a minimum weight design. The transition occurs when the station unit weight of the titanium equals the unit weight of columbium on the basis of required wall thicknesses sized by stiffness and stress requirements with a wall temperature profile extrapolated from the BAC 100-lb thrust chamber data.

The results of the thrust chamber tailoring to this application may be seen in both the weight and performance estimates for thrust levels below 500 lb, as illustrated by the 100-lb thrust chamber which exhibits over a 1% performance improvement over an equivalent thrust chamber optimized for rapid pulsing operation with a lower L* to minimize restart losses. Additional barrier cooling with fuel along the wall is used for the throttled units.

The major envelope requirements for the thrust chambers described in Figures 50 and 51 are presented in Figures 52 through 55. These are the external dimensions of the thrust chamber including the weld beads, the stiffener ribs, and the inlet fittings. An allowance for a propellant valve mounting cage is included in the chambers below 400-lb thrust. Above this range, AN type fittings are used to couple the propellant lines to the injector manifolds.

Figure 52 describes the dimensions of the combustion chamber upstream of the physical throat. These requirements are not significantly affected by the design area ratio.

Figure 53 presents the required nozzle length from the throat to the exit plane for an 80% bell nozzle. This dimension (and the exit diameter, Figure 54) is influenced by the design area ratio which establishes the performance and defines the throat area for a constant thrust. This data reflects the change in throat area with design area ratio.

Figure 55 shows the overall or total thrust chamber length required for the parametric range investigated.

The major envelope dimensions have been coded for reference on Figures 52 through 55 with the maximum diameter of the cylindrical chamber section identified as dimension C, and its length as dimension A. The nozzle maximum diameter is coded D with its length from the throat called B. The total chamber length is identified as dimension L.

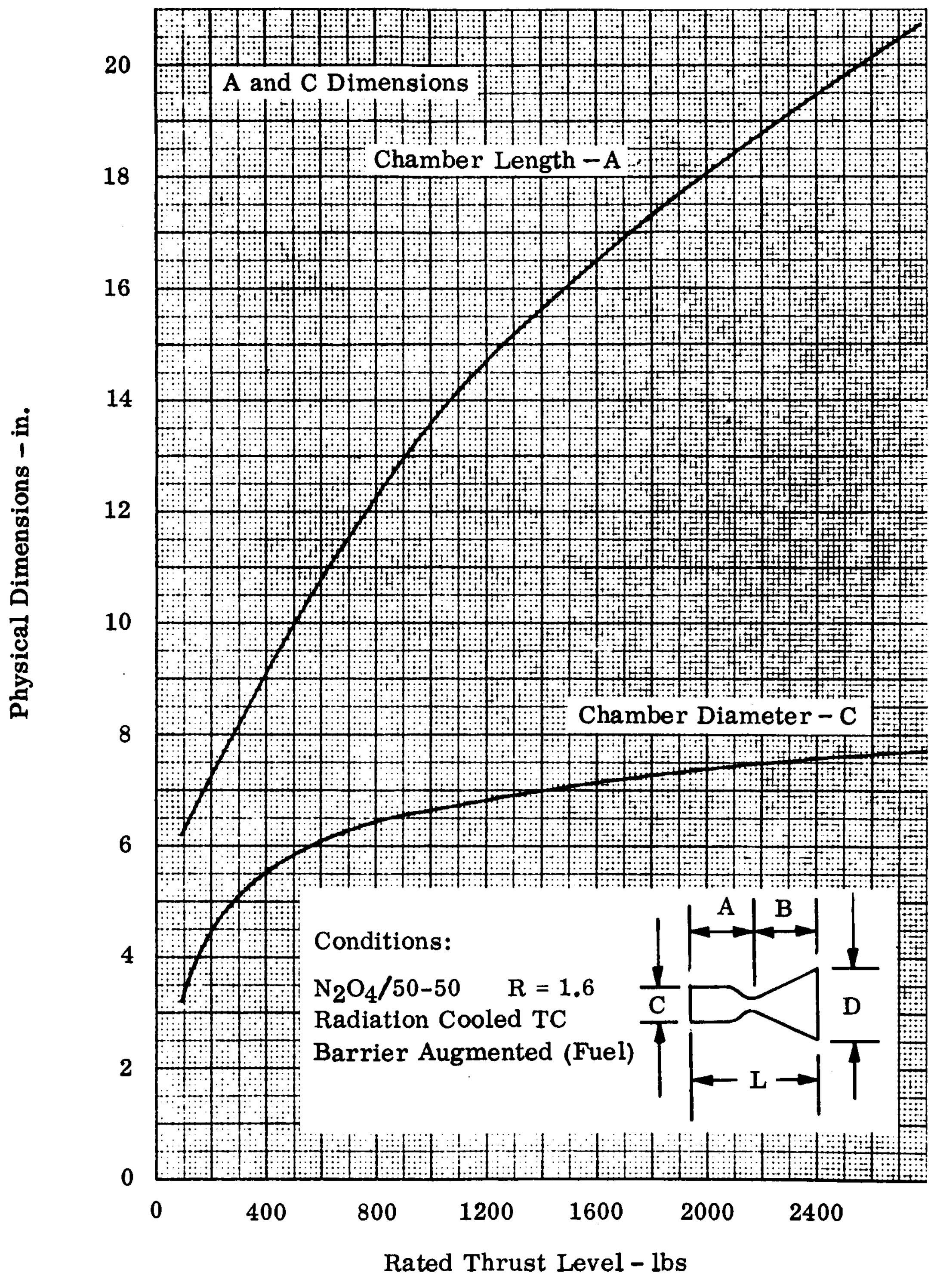


Figure 52. Thrust Chamber Envelope Combustion Chamber Length and Diameter

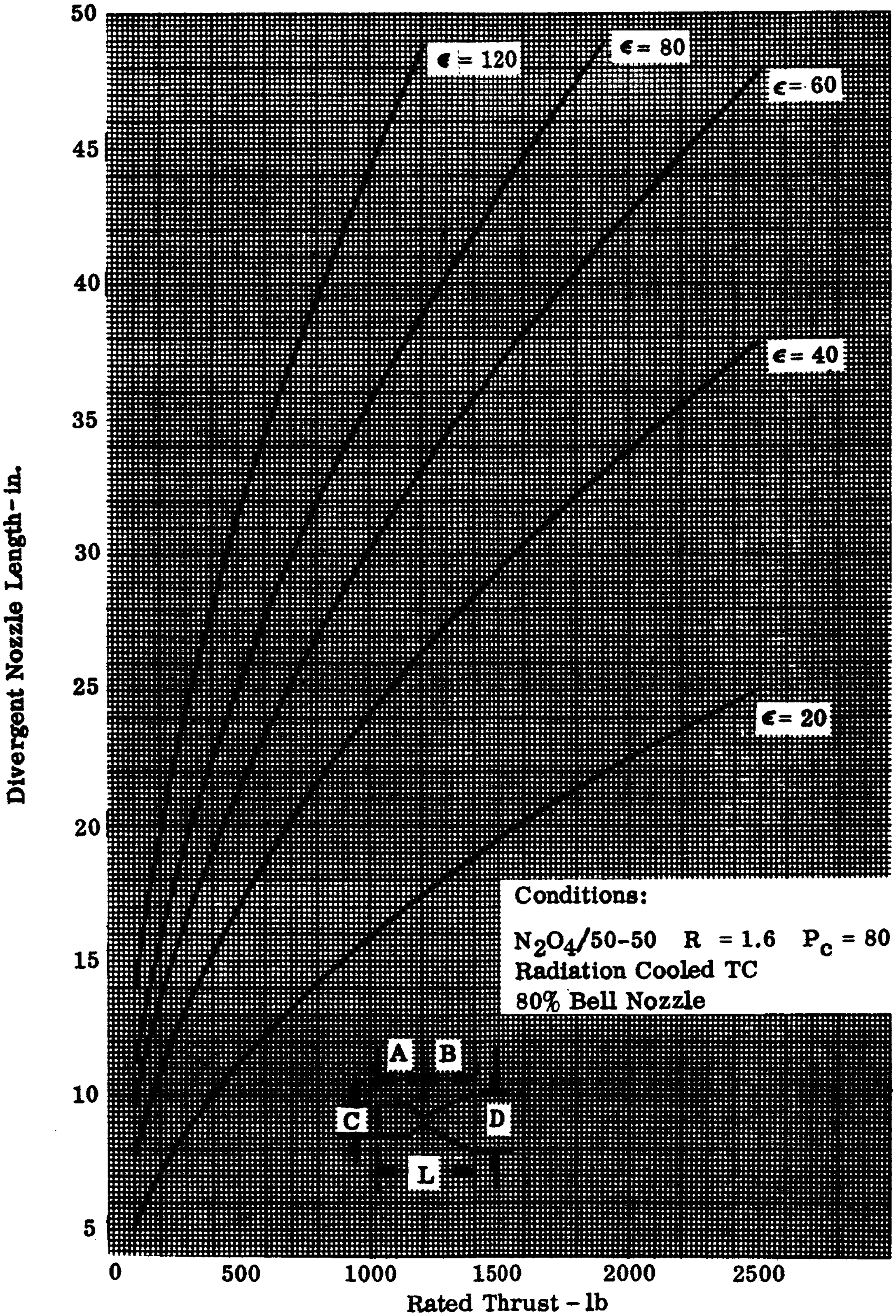


Figure 53. Nozzle Length versus Thrust and Area Ratio

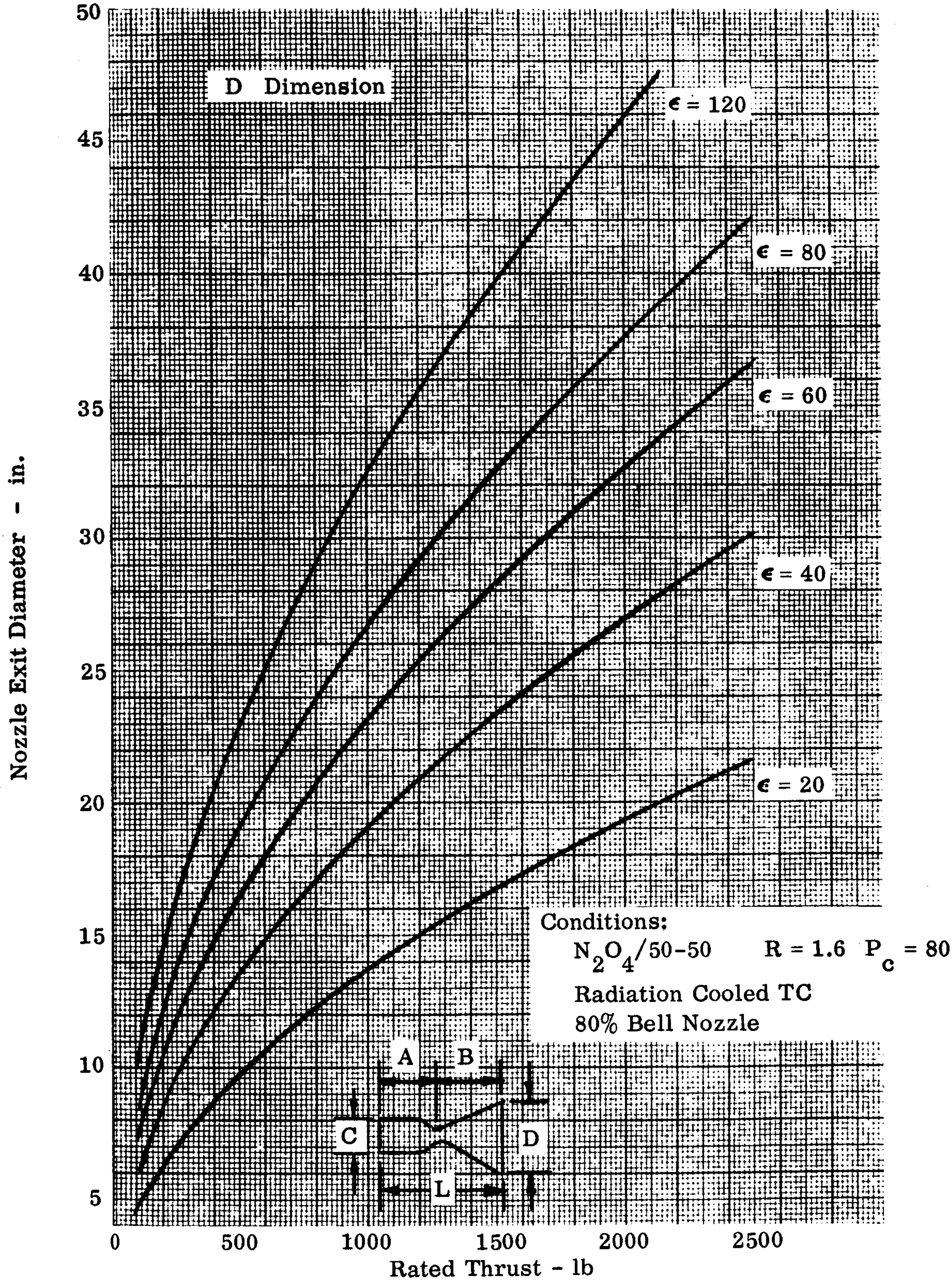


Figure 54. Nozzle Exit Diameter versus Thrust and Area Ratio

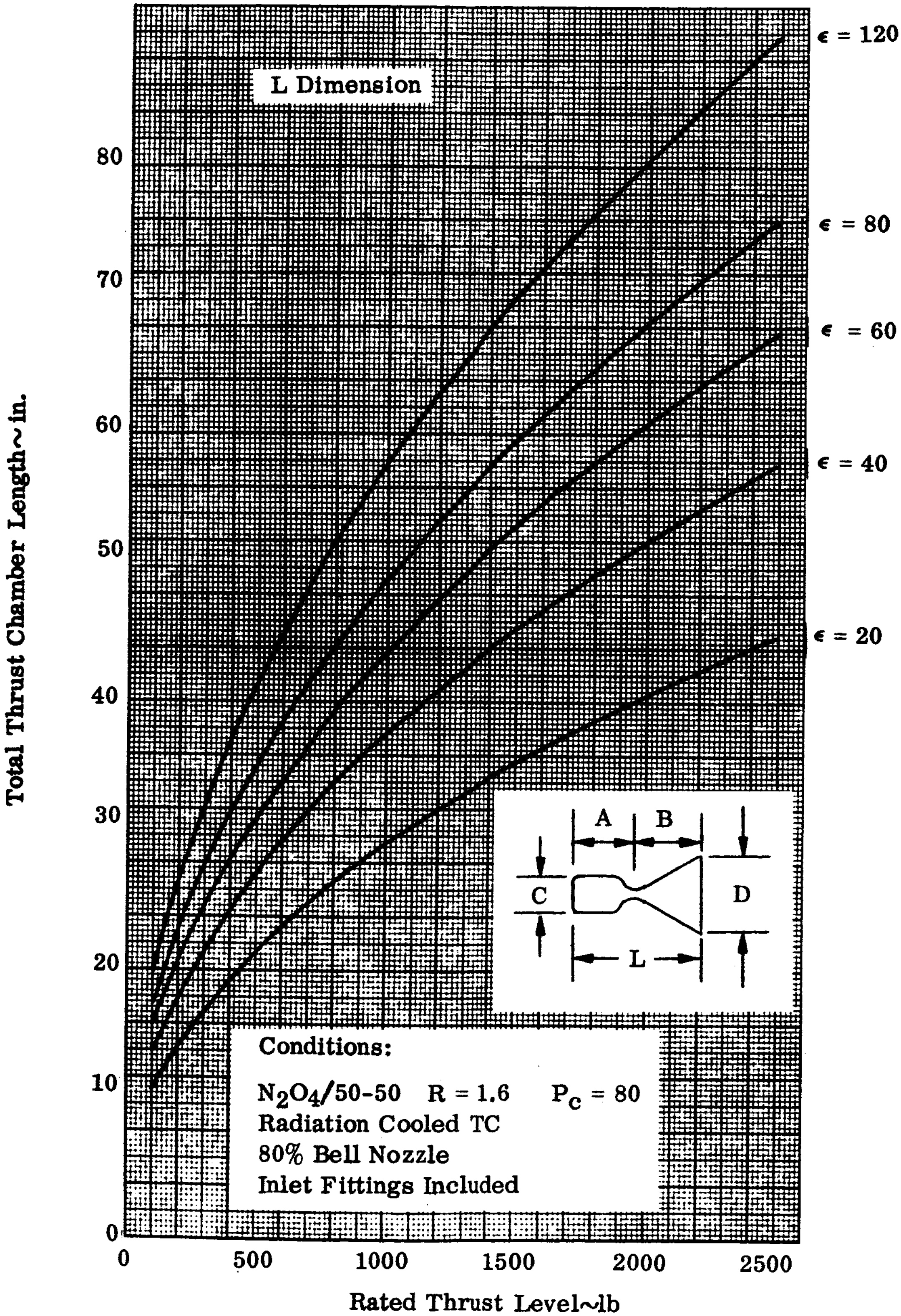


Figure 55. Total TC Length versus Thrust and Area Ratio

3. Ablative Thrust Chamber Weight and Envelope

The weight and envelope of an ablative thrust chamber has been determined for thrust levels of 100 to 2000 lb, expansion area ratios of 20 through 60, and mission burn times of 200 through 1000 seconds, using N_2O_4/50-50 propellants at a mixture ratio of 1.6 and chamber pressure of 120 psia. The LEM ascent engine thrust chamber design requirements were used for this data which corresponds directly with the escape mission requirements of this program, thus the programmed duty cycle consists of one long burn with two very short burns ($\sim$20 sec) and a 10% mission life reserve. This data is directly applicable to the transportation mission for rated thrust operation and represents only a slight conservatism for throttled operation, because the LEM design philosophy limits the operating conditions to minimize erosion induced dimensional changes with a fuel rich gas barrier adjacent to the ablative.

A word of caution is in order for ablative applications of low thrust and long burning times. Current ablative designs do not exceed 500 seconds of mission burn time (about 650 seconds total design life) however, concept feasibility has been established for 750 to 1000 seconds at the operating conditions similar to this application. The ablative data for the 1000-second burn time presented herein is an extrapolation of the 380-second LEM technology and represents a best estimate for this application. However, the developmental risks and costs will increase proportionately for any design requiring over 500 seconds mission burn time.

The subject design is for an all-ablative thrust chamber; i.e., a radiation cooled nozzle extension is not used on the grounds that an ablative chamber will be used in place of a radiation cooled chamber only where the advantage of a buried installation exceeds the limited life and increased weight penalties. Using this philosophy, a maximum skin temperature of 500°F after soakback has been used to determine the internal insulation requirements.

Figure 56 describes the weight of an ablative thrust chamber by means of a 4 parameter carpet plot. Three individual carpet plots of thrust and area ratio are shown for mission burn times of 200, 500, and 1000 seconds, respectively. These three plots are related by a linear time relationship, by lines of constant thrust (F), and area ratio (A_e/A_t) to form another carpet plot with mission burn time, as shown by the lines of increasing t_b. The darkened triangles indicate the weight of an ablative chamber of 1000 lb thrust with an area ratio of 20 for 200, 300, 400 . . . . 1000 seconds burn time. The weight of a chamber at any thrust, area ratio and burn time is obtained by entering each of the three primary carpet plots at the thrust and area ratio required. A line is drawn connecting the points thus defined, and this line is then divided into the appropriate burn time increments to define the desired chamber weight as a function of all three independent parameters (F, A_e/A_t, t_b).

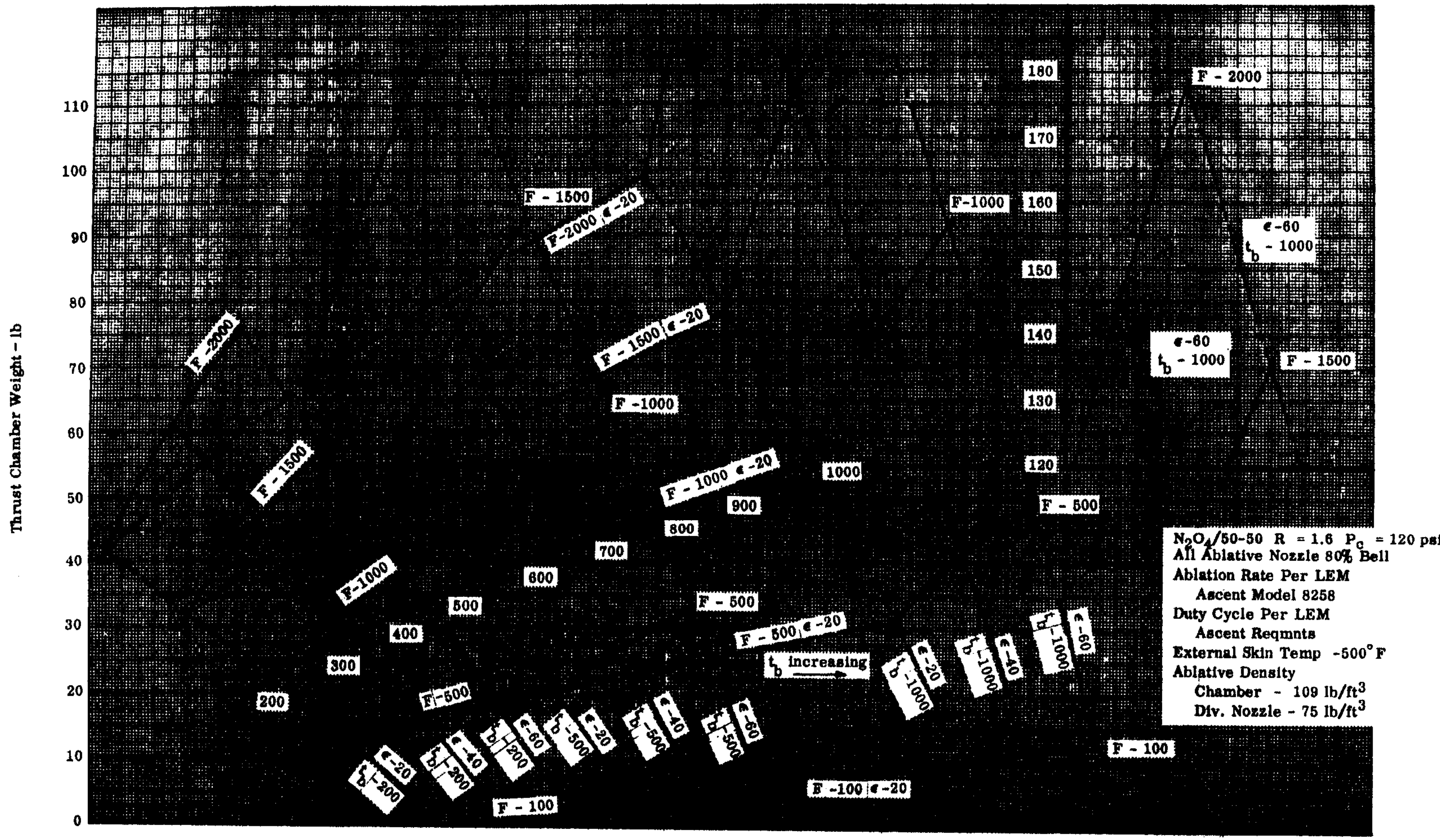


Figure 56. Ablative Thrust Chamber Weight

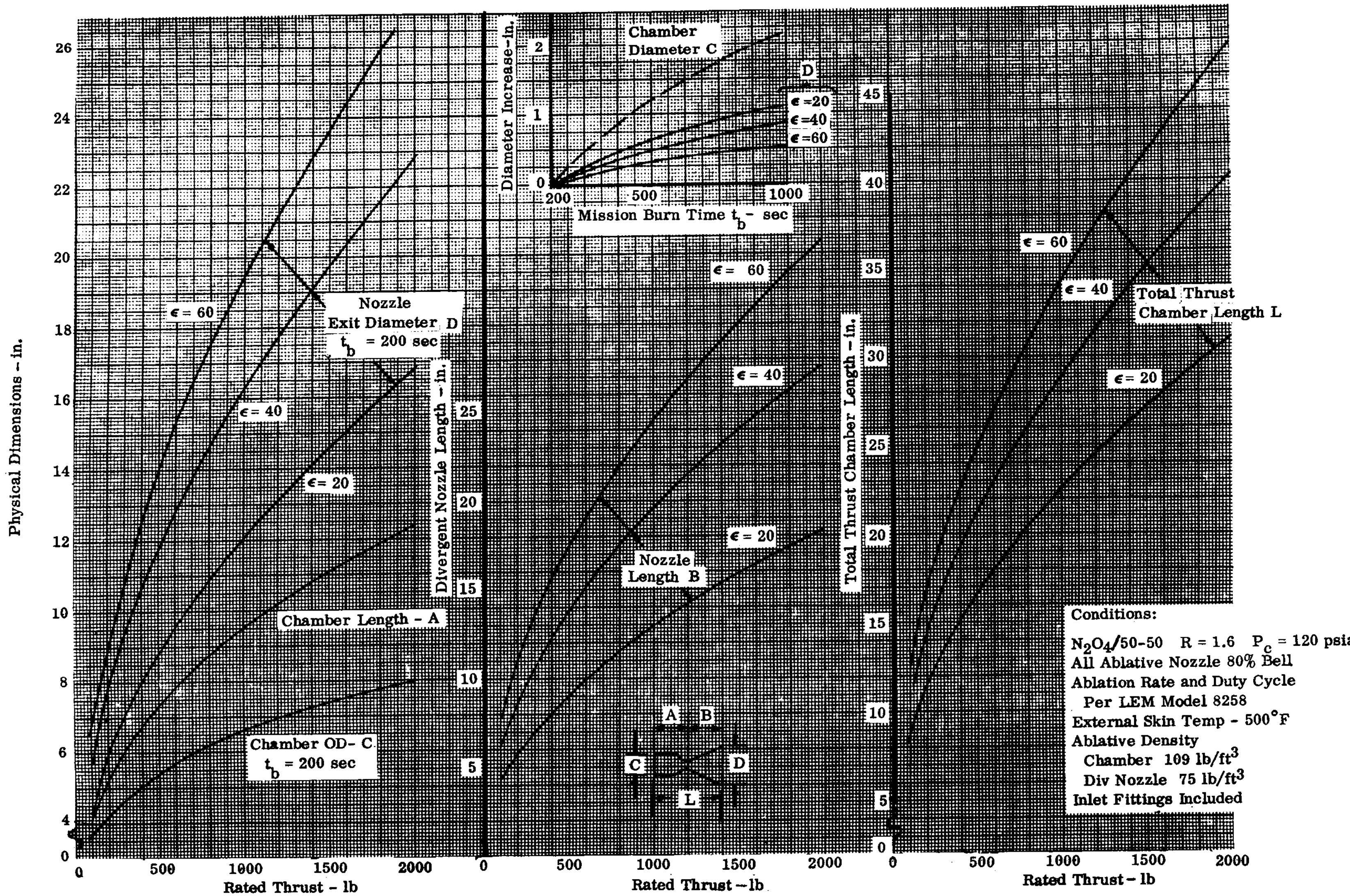


Figure 57. Ablative Duct Chamber Envelope

The envelope requirements for these chambers are given in Figure 57. The chamber length — identified as dimension "A" (see sketch in lower center of figure); the nozzle length, dimension "B"; and the total thrust chamber length, dimension "L" are independent of the mission burn time and are given as functions of area ratio and rated thrust. The chamber diameter, dimension "C", and the nozzle exit diameter, dimension "D", are plotted for a burn time of 200 seconds. The increase in these dimensions with mission burn time is shown in the insert at the top center of the figure; therefore, the total envelope is the sum of these dimensions for the desired design conditions.

E. ATTITUDE CONTROL SYSTEM

A survey of the potential attitude control propulsion systems for this application was conducted to estimate by system synthesis, the functional characteristics, component weight envelopes and operational limitations of current design technology. The survey resulted in the definition of the major system characteristics and permitted a rough comparison of candidates investigated. Inadequate definition of the attitude control system requirements and design duty cycle prevented system sizing comparisons for specific missions; however, sufficient parametric component data is provided to establish system weights and envelopes for the major candidates.

General conclusions show the advantage (simpler, less weight) of the storable bipropellant N_2O_4/0.50 H_2H_4 + 0.50 UDMH system drawing propellants from the primary propulsion system tanks. This approach provides the greatest mission flexibility by permitting propellants to be used for either primary or attitude control propulsion, increases the reliability of the system by using the standby redundancy in the primary propulsion pressurant regulation system, minimizes the number of propellants onboard the vehicle, and simplifies the propellant freezing problem during the lunar surface storage period.

The attitude control system for this application was studied in sufficient depth to enable selection of the most suitable system and to define the weight of its components as a function of duty cycle and total impulse.

An N_2O_4/50-50 system using pressurized propellants drawn from the primary propulsion system tanks is recommended as the lightest weight, least complex system of the major candidates; which included gaseous bipropellants (F_2/CH_4), monopropellants (H_2O_2, N_2H_4), and cold gas (GN_2).

Table IV is a comparison of the candidate systems for an idealized mission requirement of 8000 lb-sec total impulse for 8 thrust chambers operating in pulse mode at 0.100 sec equivalent square wave pulse width. Generalized conclusions cannot be drawn from this comparison due to the significant change in relative differences between candidates for different requirements, especially number of thrust chambers, thrust level, total impulse, and average pulse width during the mission.

TABLE IV

IDEALIZED ACS COMPARISON

	N_2O_4/50-50		H_2O_2	N_2H_4	F_2/CH_4	GN_2
	Indep.	Main Tanks				
I_{sp} at 20 lb thrust	286.1	286.1	164	240.0	340.0	68.0
% Steady State I_{sp}	.955	.955	.82	.62	.982	.982
I_{sp} delivered	273.23	273.23	134.48	148.8	333.9	66.8
Mixture Ratio (R)	1.6	1.6			3.0	
Weight Estimates (lb)	29.88				24.0	
Prop. Load Fuel	11.49	11.49	60.70	54.86	6.0	(119.8)
Ox	18.39	18.39			17.971	
Prop. Tank Fuel	2.35	(2.26)**	11.15	5.7	20.1***	313.9***
Ox	2.85				40.3***	
Pressurant & Tank	1.53		8.35	5.0	-	-
Components*	9.0	0.6	5.7	5.7	11.1	5.9
Thrust Chamber Unit	2.05	2.05	1.45	1.45	4.3	0.90
(8 TCA) Total	16.40	16.40	11.60	11.60	34.0	7.2
ACS Weight Total	62.0	49.1	97.5	89.9	105.5	327.0

I_T = 8000 lb-sec at 0.100 equivalent square wave - hot bed performance

* Component weight does not include line weight estimate

Expulsion efficiency of liquid systems = 98%

** Equivalent increase in main tank weight to store and pressurize 30 lb of propellant for one man vehicle.

*** Includes propellant, residuals and tanks.

A description of the candidate systems as employed for this application follows. The delivered steady state performance of these systems is shown in Figure 58 for the design thrust size, with the degradation incurred by pulsing described as the ratio of delivered specific impulse to the steady state value in Figure 59.

1. Liquid Bipropellant System (N_2O_4/0.50 N_2H_4 + 0.50 UDMH)

The conventional regulated pressurized propellant feed system using N_2O_4/0.50 N_2H_4 and 0.50 UDMH as propellants is shown schematically in Figure 60 It includes an estimate of the major component weights for this system which is independent of the main propulsion system. Its major advantages consist of high specific impulse with fast transient response (reproducible impulse bits down to 7 milliseconds) and commonality of propellants throughout the lunar vehicle. Thrust levels from 10 lb up are available, using propellants drawn from the main tanks when operating at a mixture ratio of 1.6, without affecting mission flexibility. Thrust levels below 10 lb down to 1 lb are feasible at richer mixture ratios ($R \approx 1.4$); however, these thrust levels below 5 lb are not recommended for this application due to problems of injector clogging and the dual capacity requirements of the regulator feeding both the main propulsion and the ACS units. Other liquid propellants can be used, but are not attractive for this mission when compared to the performance and simplicity obtained by using the primary propulsion propellants. Selection of main tank feed minimizes the freezing problem by providing the bulk storage heat sink during the period when the vehicle is on the lunar surface.

2. Liquid Monopropellant Systems (N_2H_4, H_2O_2)

Two monopropellants were considered for this application, anhydrous hydrazine and 90% peroxide. The peroxide system is developed and flight proven while the hydrazine system using the Shell catalyst is in the early stages of development. The hydrazine system is completely storable, while the peroxide system is significantly degraded by a high temperature ($> 90^\circ$ F) storage environment.

A representative hydrogen peroxide system for the ACS propulsion requirements has been investigated for comparison with the other PPD candidates. Decomposition of the propellant is accomplished by a catalyst bed contained within the thrust chamber. All components of a similar propulsion system have been developed and man rated for the Mercury program, which promises a minimum of development risks and therefore costs, as compared to the other candidates. Anticipated changes required to adapt the Mercury technology are limited to component configuration changes. This system offers other advantages, including minimum unit production costs, adiabatic thrust chamber operation (buried installation possible), and relatively low temperature, transparent (visual and at radar/radio frequencies) exhausts. Its advanced development status promises the highest reliability of any of the candidates for low development funding levels.

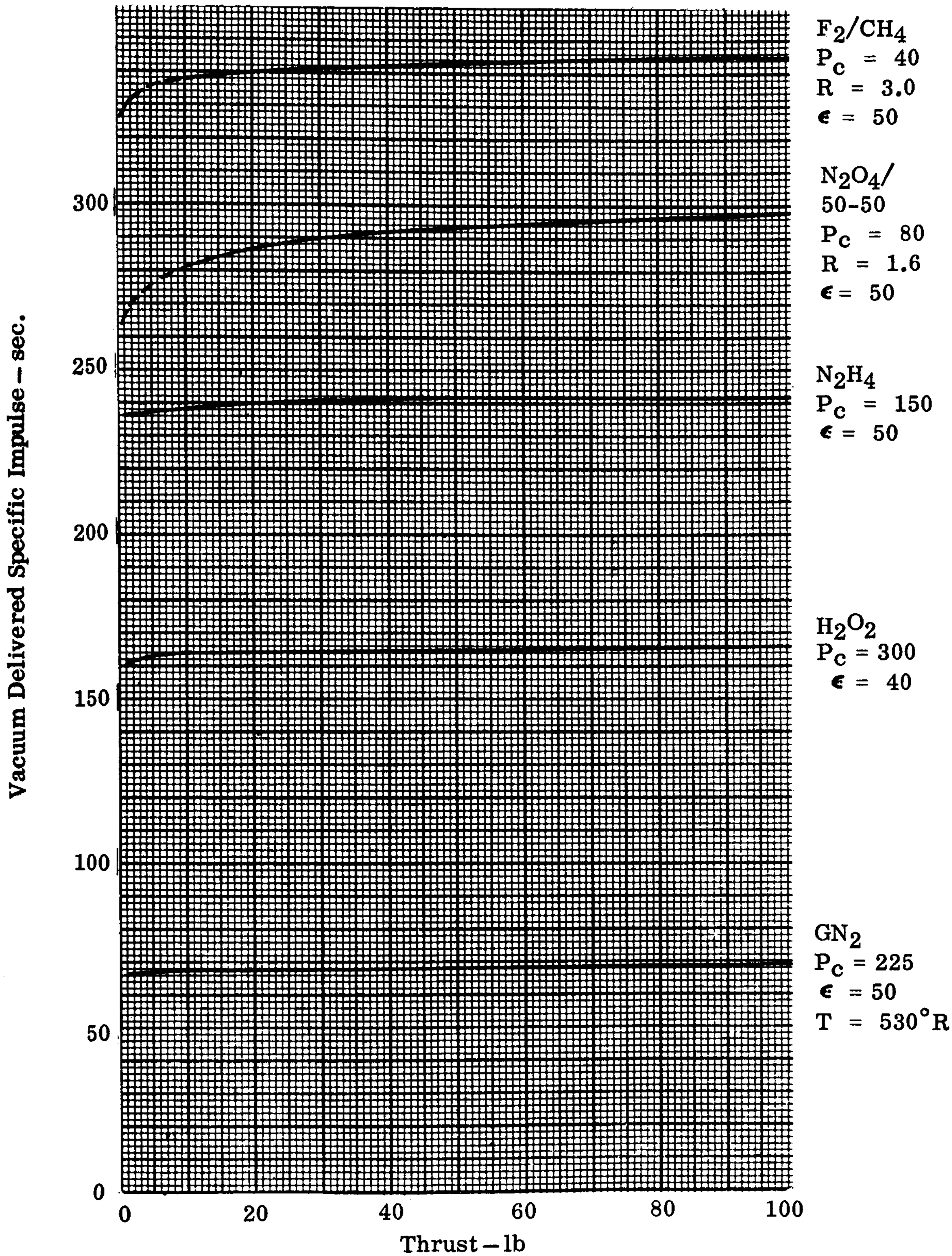


Figure 58. Steady State I_{sp} versus Thrust

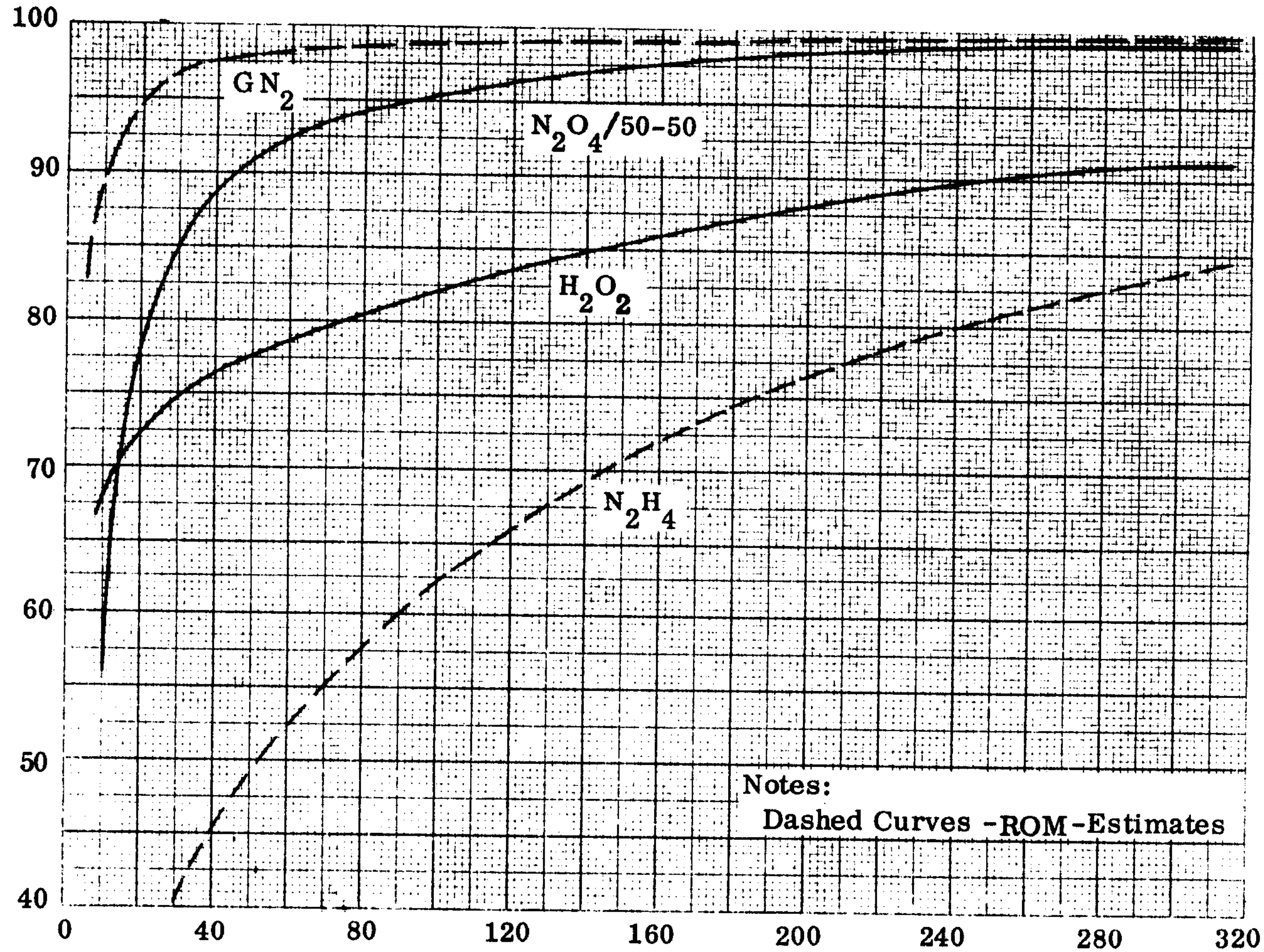
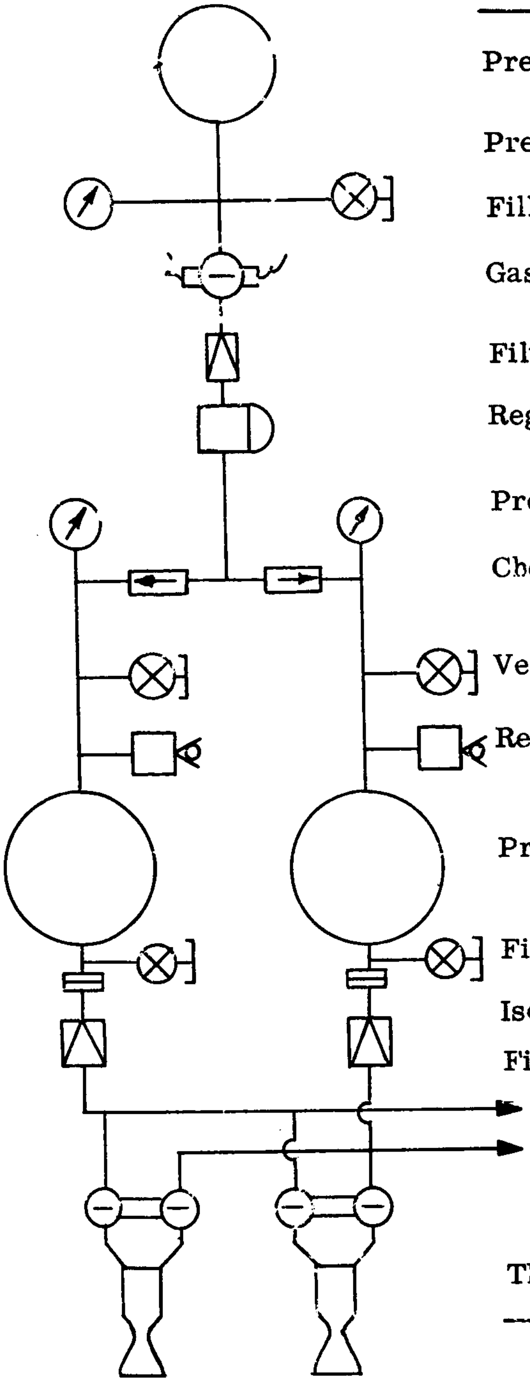


Figure 59. Equivalent Square Wave Impulse Bit Width Milliseconds



System Design Thrust Level	Component Weight Estimate 10 ⟶ 100
Pressurant and Tank	Figure 61
Pressure Gage	1.2
Fill and Vent Valve	0.1
Gas Start Valve	0.3
Filter	0.1
Regulator	0.9
Pressure Gage	2.4
Check Valve	1.0
Vent Valve	0.4
Relief Valve	1.6
Propellant Tank	Figure 61
Fill and Drain Valve	0.4
Isolation Burst Discs	0.4
Filter	0.2
Thrust Chamber Assembly	Figure 62
Total Fixed Weight	9.0

Figure 60. ACS Liquid Bipropellant System Schematic

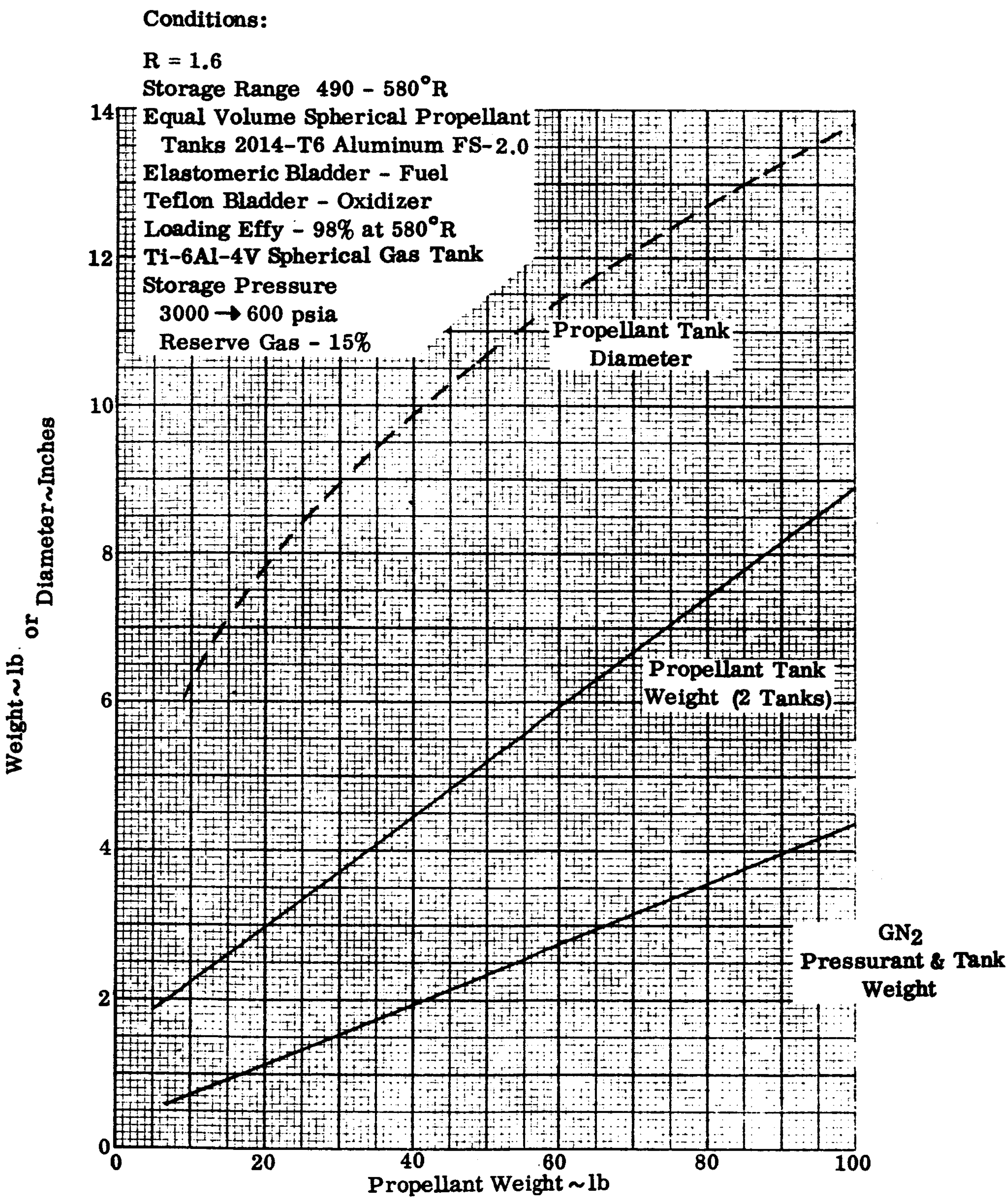


Figure 61. ACS N_2O_4/0.5 N_2H_4 + 0.5 UDMH Propellant Tank Weight and Diameter GN_2 Pressurant and Tank Weight

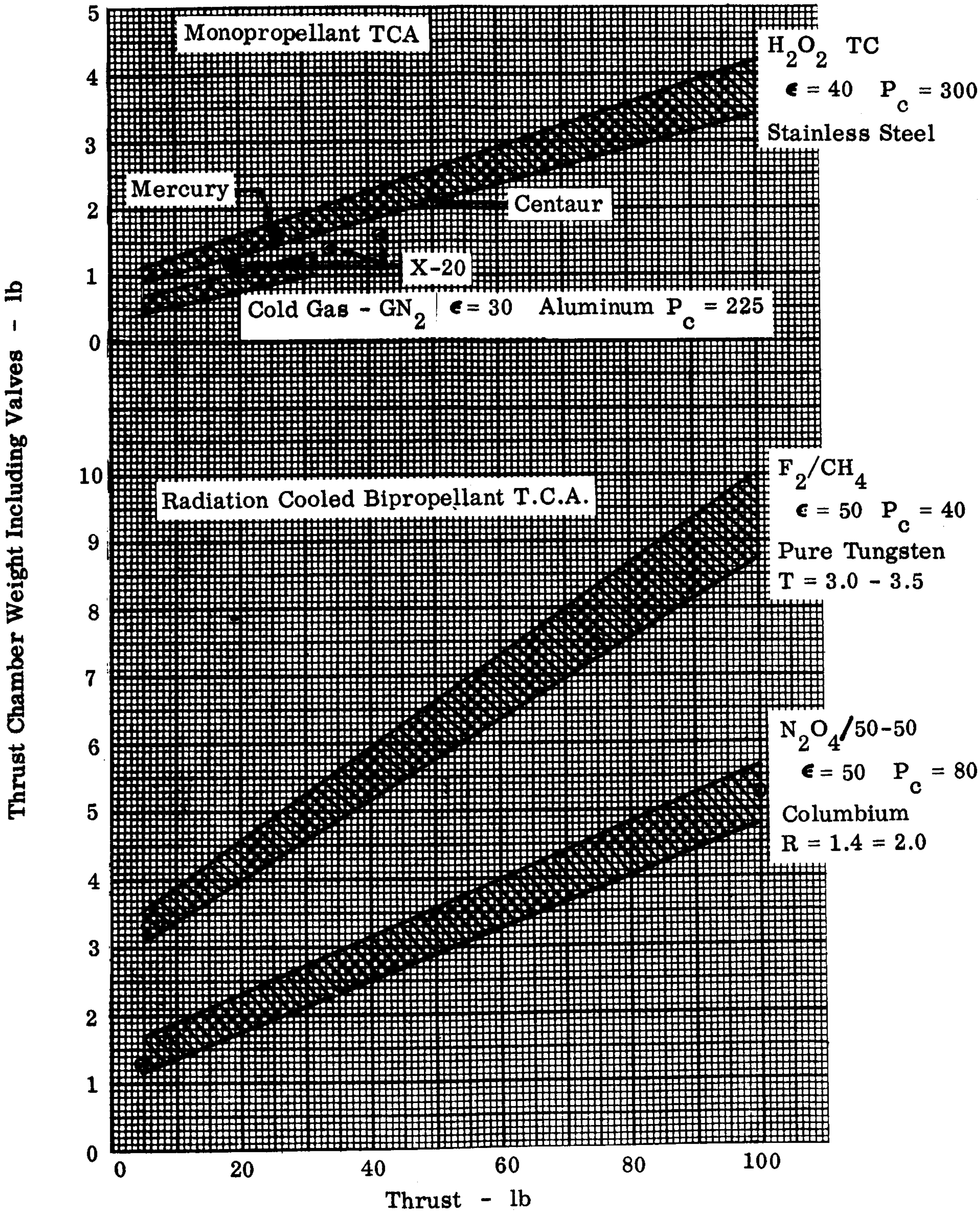


Figure 62. TCA Weight versus Thrust

The low performance level (I_{sp} of 150 to 160 sec) and the slow decomposition of this propellant when stored at the upper environmental temperatures expected on the lunar surface provide the major problems in the use of hydrogen peroxide. The storage problem has been minimized in the accompanying system by using a bladder of silastic S-9711 (a material shown to provide a minimum of decomposition) and by minimizing the wetted surface known to cause decomposition. These effects are shown in Figure 63 for a peroxide system with 15% ullage, a compromise of surface to volume ratio, and pressure rise (due to gas evoluation from decomposition) selected for this study. Note that increasing the ullage volume decreases the tank pressure, but increases the wetted surface for the same propellant volume which results in an increased decomposition rate with lower performance, and increases the quantity of gas trapped on the propellant side of the bladder — a potential problem in terms of propellant acquisition under zero g conditions. A representative system schematic is shown in Figure 64. Note the start valve is mounted directly to the propellant tank.

The response times and performance level of a peroxide system are a function of the bed temperature and, therefore, are readily influenced by the duty cycle (Reference Figure 67). The pulsing characteristics of the peroxide chamber are good (approximately a 20 millisecond repeatable equivalent square wave pulse width), but are also influenced by the bed temperature. Examination of the probable mission duty cycle discloses no significant compromises caused by these characteristics and therefore represent only a weight penalty due to the performance degradation.

A first pass, preliminary estimate of a representative hydrazine monopropellant system for the ACS propulsion requirements has been investigated to permit comparison with the other candidates. Decomposition of the hydrazine is accomplished by thermal and catalytic decomposition initiated by a catalyst developed by Shell for NASA. The proposed system is shown schematically in Figure 64. This system offers the highest performing stable monopropellant currently available with the reliability advantages incurred by a single propellant in the feed system. The relatively low gas temperature (2000°F) offers another advantage in permitting buried installation, operating adiabatically, using current structural materials.

Anticipated problems stem primarily from the nature of the system using a catalyst bed for ignition and from the present low level of development and lack of design technology for using the Shell catalyst. Neither problem is sufficient to reject the hydrazine monopropellant from consideration for this application. Current reported test data is sufficient to indicate reliable ignition can be assured and a performance level of 240 seconds specific impulse in steady state operation can be reasonably expected.

The pulsing characteristics appear to be the foremost problem area with this system. There is no reported test data for short pulses (below 100 ms) in a hydrazine monopropellant system. An order of magnitude estimate of the pulsing characteristics was attempted to provide a guide for integration of the system.

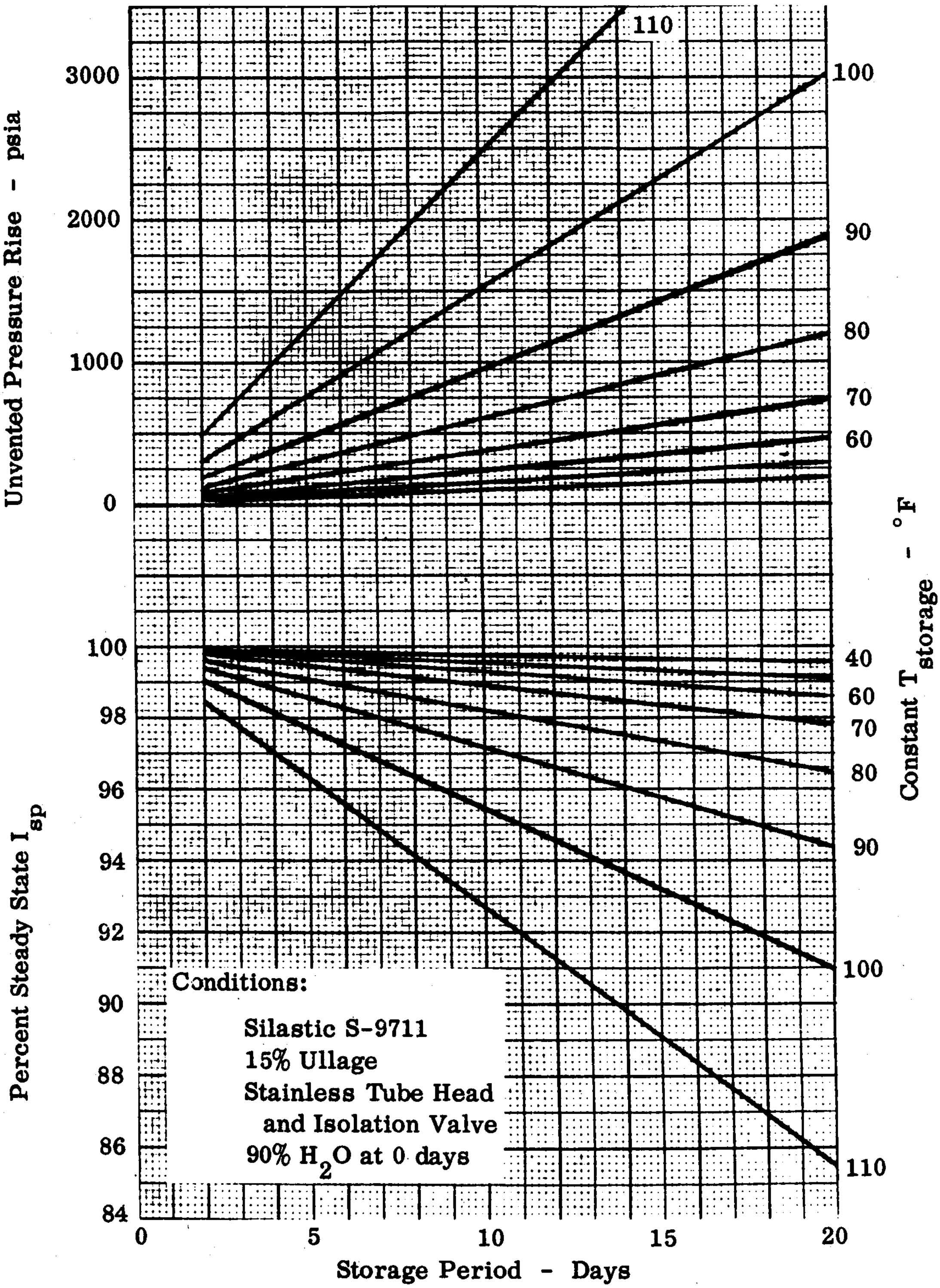
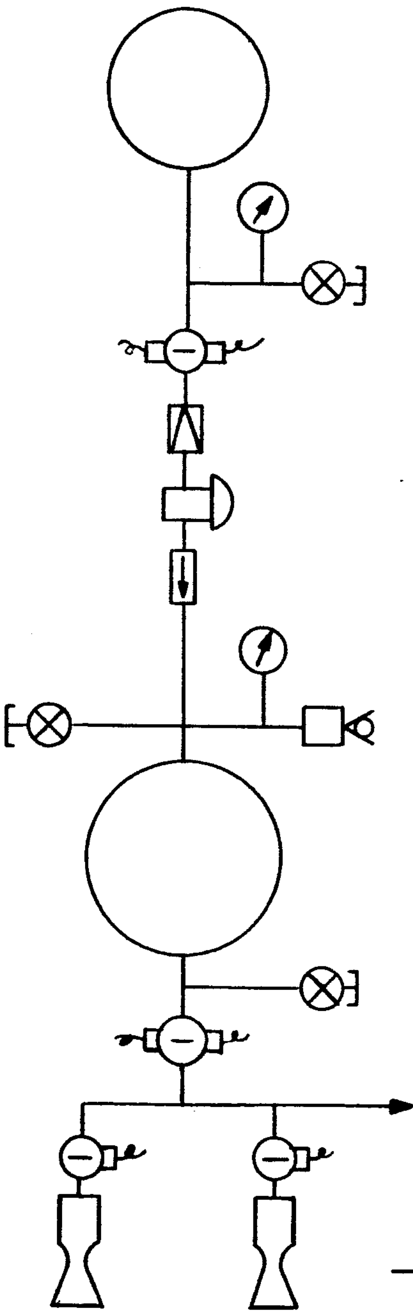


Figure 63. H_2O_2 Pressure Rise and Performance versus Storage Period



System Design Thrust Level	Component Weight Estimate 10 → 100
Pressurant and Tank (H_2O_2 System)	Figure 65
Pressurant and Tank (N_2H_4 System)	Figure 66
Pressure Gage	1.2
Fill and Vent Valve	0.1
Gas Start Valve	0.3
Filter	0.1
Regulator	1.4
Check Valve	0.1
Pressure Gage	1.2
Vent Valve	0.1
Relief Valve	0.6
Propellant Tank (H_2O_2)	Figure 65
Propellant Tank (N_2H_4)	Figure 66
Fill and Drain Valve	0.3
Isolation Valve	0.3
Thrust Chamber Assembly	Figure 62
Total Fixed Weight	5.7

Figure 64. ACS Monopropellant System Schematic N_2H_4 and H_2O_2

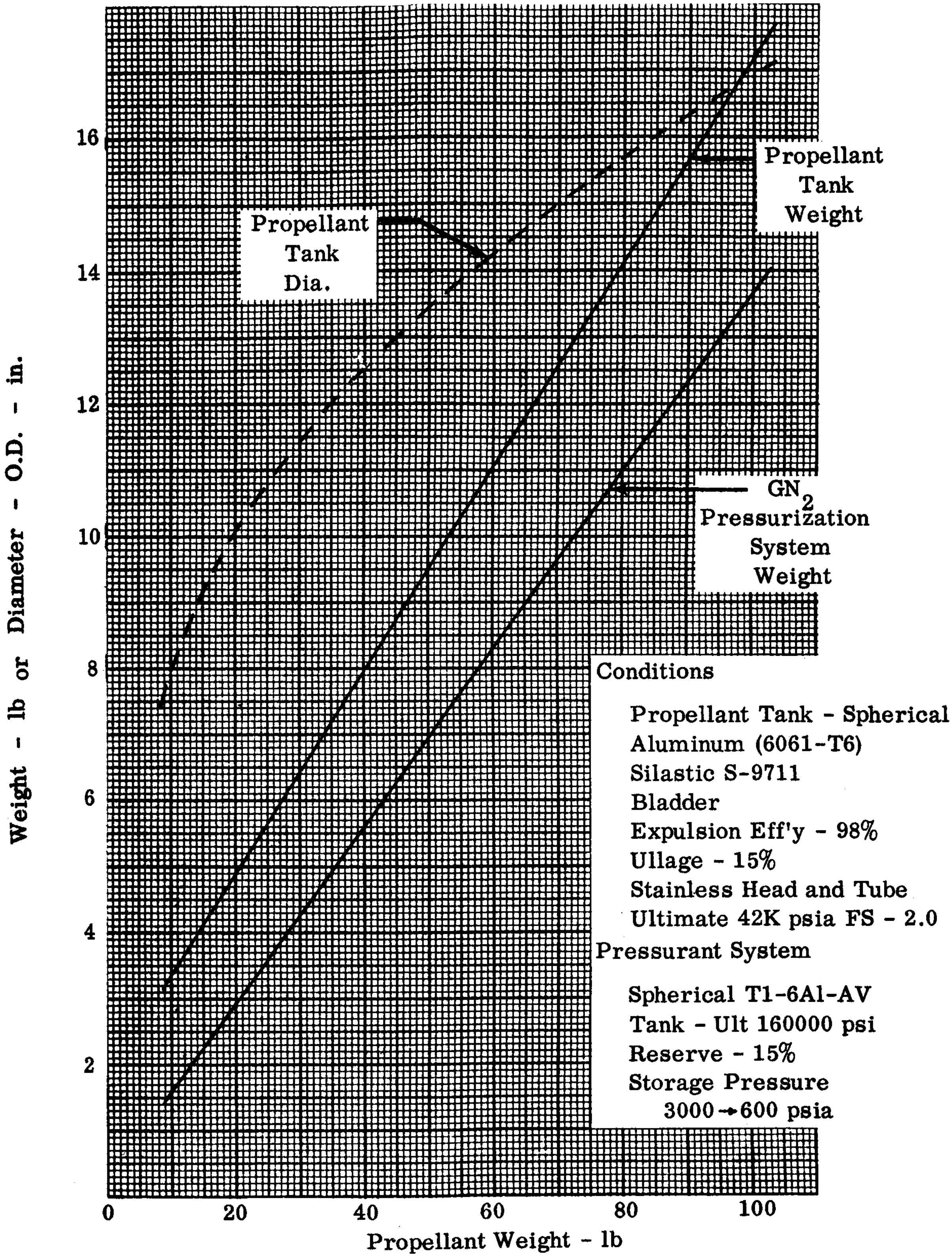


Figure 65. ACS H_2O_2 Propellant Tank Weight & Diameter GN_2 Pressurant and Tank Weight

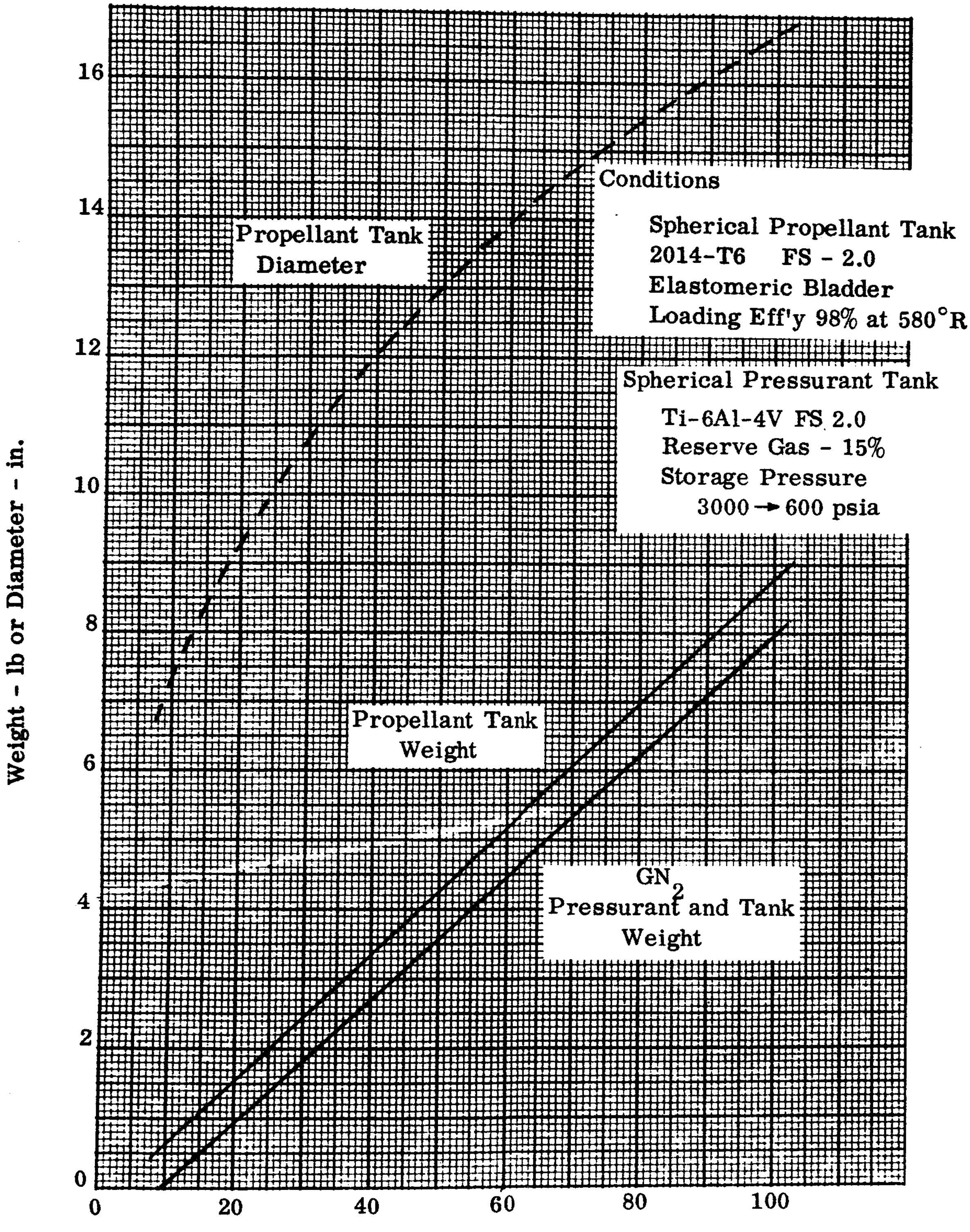


Figure 66. ACS Hydrazine (N_2H_4) Propellant Tank Weight and Diameter
GN_2 Pressurant and Tank Weight

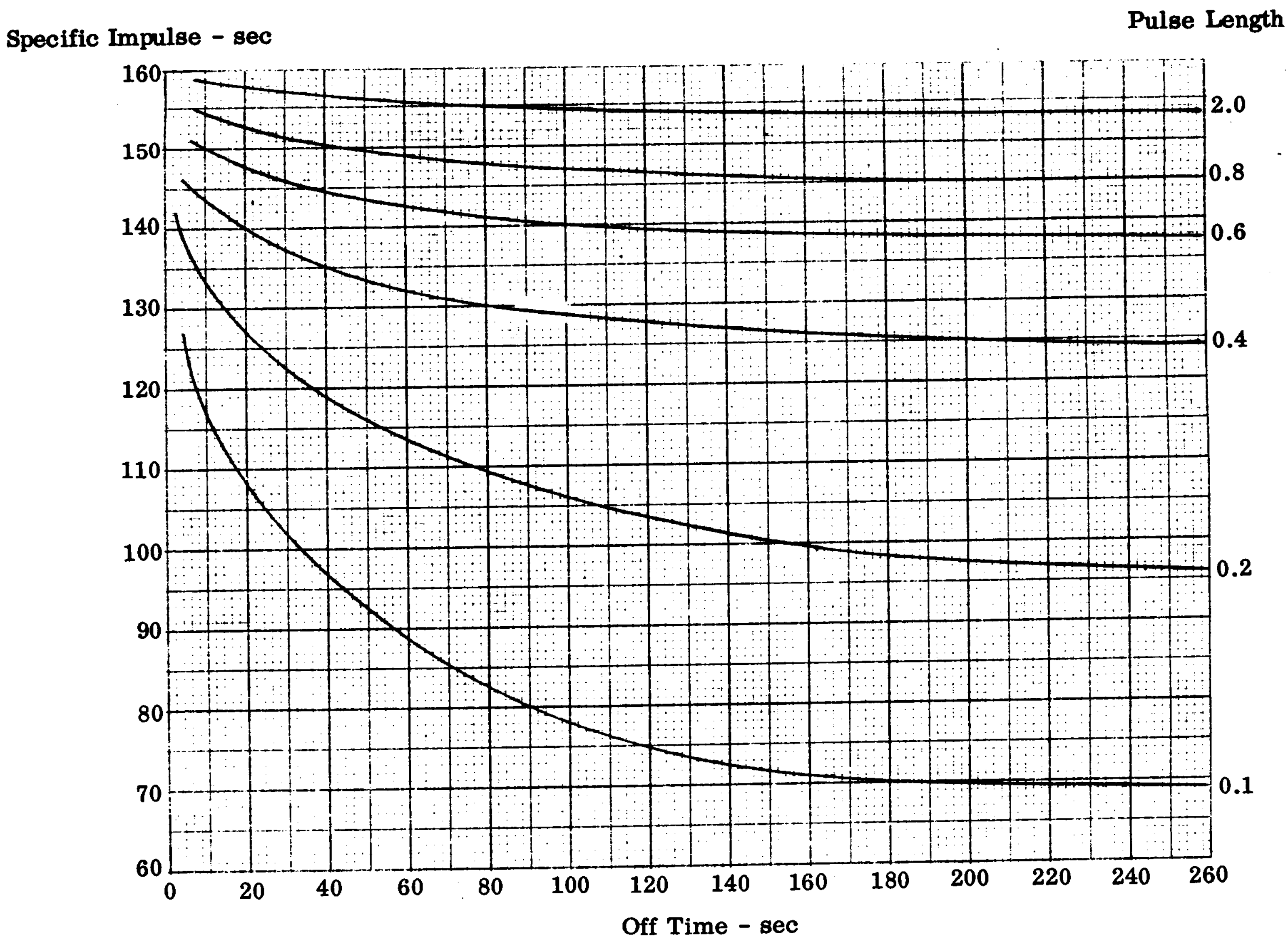
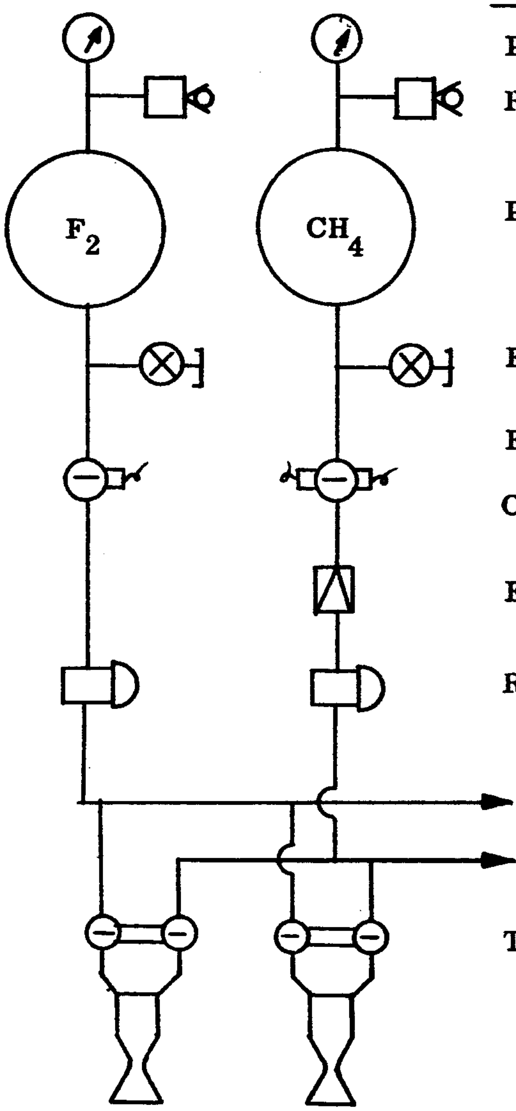


Figure 67. Specific Impulse versus Off Time for Various Pulse Length for H_2O_2



System Design Thrust Level	Component Weight Estimate 10 ⟶ 100
Pressure Gage	3.5
Relief Valve	2.0
Propellant Tank	Figure 69
Fill and Vent Valve	0.6
Fuel Isolation Valve	0.3
Oxidizer Isolation Solenoid	0.6
Fuel Filter	0.1
Regulator	4.0
Thrust Chamber Assembly	Figure 62
Total Fixed Weight	11.1

Figure 68. ACS Gaseous Bipropellant System Schematic F_2/CH_4

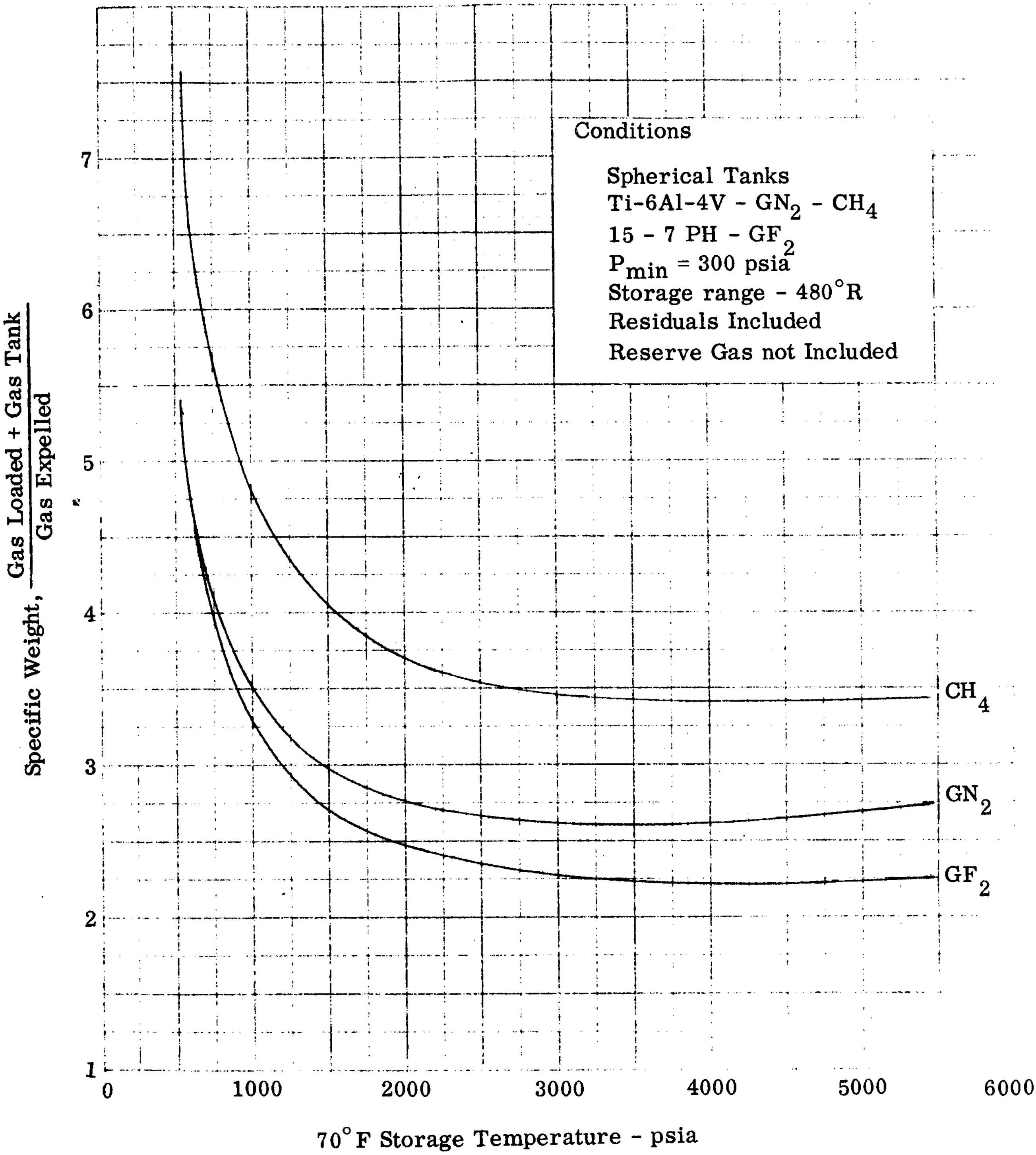
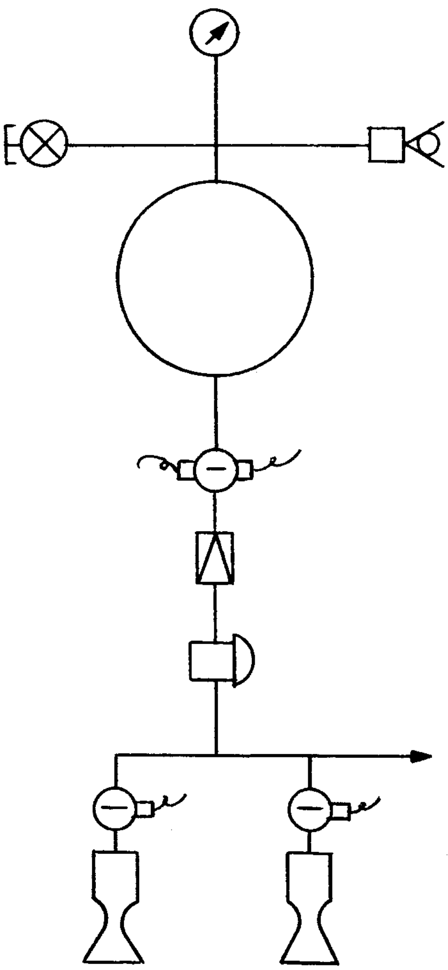


Figure 69. Specific Gas Storage Weight versus Storage Pressure



System Design Thrust Level	Component Weight Estimate 10 → 50
Pressure Gage	1.2
Fill and Vent Valve	0.1
Relief Valve	0.8
Propellant Tank	Figure 69
Isolation Valve	0.3
Filter	0.1
High Capacity Regulator	3.4
Thrust Chamber Assembly	Figure 62
Total Fixed Weight	5.9

Figure 70. ACS Cold Gas System Schematic GN_2

The initial temperature of the catalyst bed is found to have a significant effect on the response times for the system, but does not affect the delivered performance level I_{sp} (based on BAC test experience). Repeatability of the impulse bit is also affected by the response times and can present a problem when short pulses are required before the bed has reached operating temperature. The response times of the system using a hot catalyst bed may, in the future, approach a minimum of 40 milliseconds, apparently a realistic limit for this system. Reduction of this start time without flooding the bed, by employing an overside bed, increases the chamber weight, increases the quantity of propellant contained in the chamber at shutdown, and may decrease the performance level by inducing greater dissociation of ammonia.

3. Gaseous Bipropellant Systems (F_2/CH_4)

A representative gaseous bipropellant system using fluorine and methane propellants presents the highest specific impulse of the candidate systems with wide storability (-258 to 600°F). In addition, the system is relatively simple, as shown schematically in Figure 68. This system does not appear attractive for this application due to the high temperature toxic exhaust products, the low level of design technology, and the system weight disadvantage in the anticipated total impulse range.

4. Cold Gas Blowdown System (GN_2)

The cold gas system offers the least complex candidate studied for this application, as shown schematically in Figure 70. Although simple, reliable, and inexpensive, this system is also the heaviest candidate with the largest envelope requirements in the total impulse range expected for this vehicle (4 to 10K lb-sec) and is not recommended for this application. Similar systems which heat the gas or use other propellants such as isobutane are still not competitive in this impulse range.

SECTION IV

TANKS AND BLADDERS

A. TANKAGE

1. System Requirements

For satisfactory system operation in a zero gravity environment, provisions must be made in the propellant feed system to ensure propellant delivery to the engines at all times and to prevent gas ingestion by the engines in quantities which could induce erratic engine operation.

While all vehicles may not be subjected to trajectories which induce a zero gravity condition, complete mission flexibility can only be assured by incorporation of positive expulsion devices in the propellant feed system to enable hover as well as ballistic trajectory missions to be flown, so that the most efficient means of transportation may be exploited.

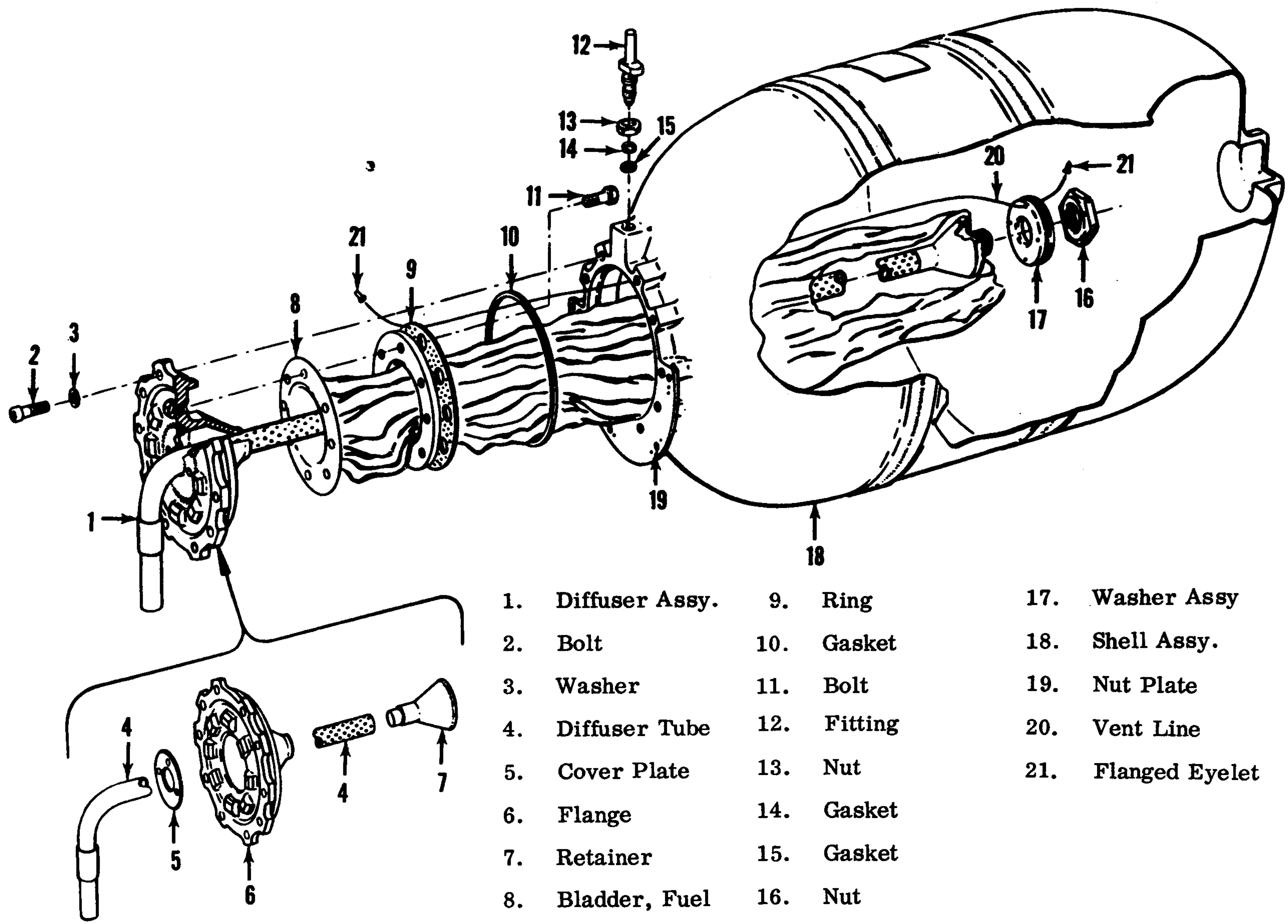
2. Positive Expulsion Devices

Several means of providing positive expulsion have been considered for use in the personnel propulsion devices. The systems were evaluated based on weight penalty, expulsion efficiency, development status and special contribution to the overall vehicle performance. Table V presents a comparison of the various expulsion systems.

a. Bladder

A bladder is a flexible membrane which is installed within the tank shell to maintain separation between the propellant and pressurant when the vehicle is subjected to zero gravity. The propellant is contained within the bladder and is expelled when the bladder collapses around a diametrically located diffuser tube. This action conducts the propellant to the outlet port of the tank when pressurant is introduced between the tank wall and the bladder. Bladders provide wide flexibility in design. Their applicability has been demonstrated with both spherical and cylindrical tanks. It is possible to install and operate the tanks with the diffuser tube either in the horizontal or vertical position, however, vertical installation is preferable to prevent the bladder from plugging the outlet holes on the diffuser tube. Figure 71 presents a typical bladder installation for use in zero g environment.

Bladder Materials - Silicone rubber bladders have successfully been used on 90% hydrogen peroxide tanks requiring up to 200 expulsion cycles. The material designation is (S-9711) and the bladder thickness is approximately 0.040 inches.



1.	Diffuser Assy.	9.	Ring	17.	Washer Assy
2.	Bolt	10.	Gasket	18.	Shell Assy.
3.	Washer	11.	Bolt	19.	Nut Plate
4.	Diffuser Tube	12.	Fitting	20.	Vent Line
5.	Cover Plate	13.	Nut	21.	Flanged Eyelet
6.	Flange	14.	Gasket		
7.	Retainer	15.	Gasket		
8.	Bladder, Fuel	16.	Nut		

Figure 71. Bladder Installation

TABLE V

EXPULSION SYSTEM COMPARISON

Expulsion System	Relative Weight	Expulsion Efficiency (%)	Development Status	Remarks
Bladder	Light	98	Operational for H_2O_2. Undergoing development testing for N_2O_4/50-50 expected to be operational for Apollo/LEM	Low dynamic restraint to propellant slosh modes. Reusable (50 expulsion cycles)
Bellows	Heavy	94	Operational	Applicable to cylindrical tanks only. Reusable
Convoluted Diaphragm	Heavy	96-98	Development - successfully subjected to expulsion and dynamic tests	Provide dynamic restraint to propellant slosh - good c.g. control single mission life
Surface Forces	Light	90	Experimental	Provide no propellant damping - highly reliable
Spin Expulsion	Heavy	90	Experimental	Burdened by dynamic seals for extended time in space. Power required for spin.

For positive expulsion of N_2O_4/50-50 N_2H_4 UDMH propellants, a 0.006 inch teflon bladder has successfully been used in the Agena secondary propulsion system. Development tests are currently in progress for the Apollo command module secondary propulsion employing similar principles. Fifty expulsion cycles have been demonstrated to date with teflon bladders.

b. Bellows

Bellows may be installed within a cylindrical tank to provide positive expulsion. In order to improve the expulsion efficiency of this device the propellant must be contained within the bellows and expelled by a collapsing action of the bellows induced by insertion of the pressurant within the tank. Propellant tanks containing bellows for positive expulsion result in larger weight per unit volume because of the following characteristics.

(1) Tank geometry is limited to cylindrical shapes.

(2) The effective loading volume of the tank is defined by the mean diameter of the bellows rather than the inside diameter of the tank.

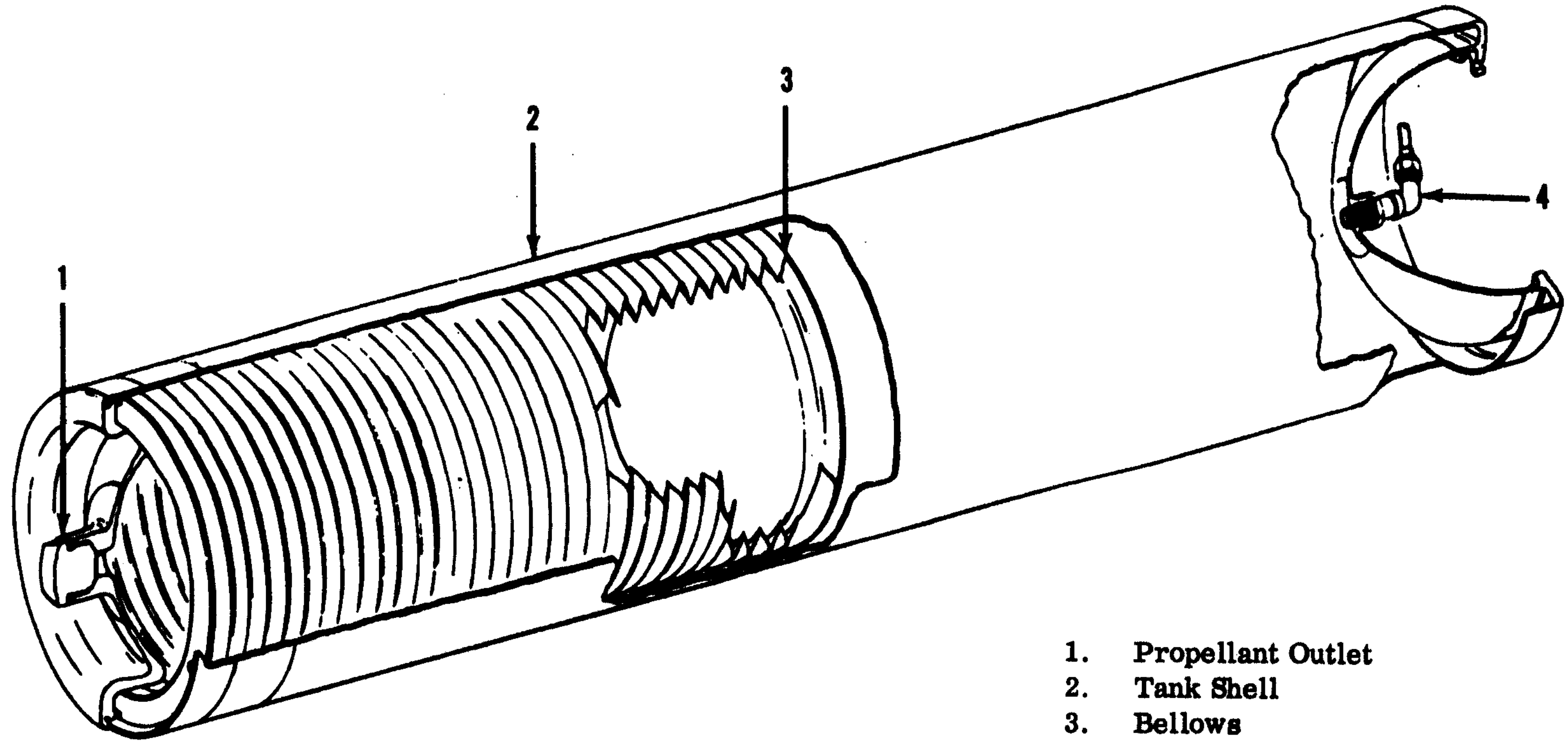
(3) In order to reduce the residual,propellants tank closures must assume shapes which deviate from optimum.

(4) Surface area of bellows is much greater than that of a bladder-type expulsion device while the thickness of the bellows and bladders is approximately equal, and the density of the bellows material is greater than the density of the bladder material.

In general, bellows represent an acceptable positive expulsion device only when material compatibility with the propellants cannot be attained with bladder materials. A typical positive expulsion tank employing bellows is shown in Figure 72.

c. Convoluted Diaphragms

Current development programs concerned with bladder material compatibility with active propellants have yielded a new concept in positive expulsion techniques. This concept is shown in Figure 73 at various stages of expulsion. Positive expulsion is provided by a double convoluted metallic diaphragm expanding against the wall of the tank. A desirable by product of this concept is its ability to eliminate center of gravity excursions by accurate, symmetrical positioning of the propellants remaining in the tank between the diaphragm and tank wall. Propellant slosh is also restrained by the rigidity of the diaphragm, thus eliminating the vehicle dynamic uncertainty caused by propellant shifting at the initiation of commands. This metallic diaphragm concept has been successfully demonstrated by Bell Aerosystems Company with regard to expulsion and dynamic tests under an Air Force contract. The main disadvantage of this system is its inability to comply with the multiple mission requirement of the personnel propulsion devices.



1. Propellant Outlet
2. Tank Shell
3. Bellows
4. Gas Inlet

Figure 72. Typical Positive Expulsion Tank

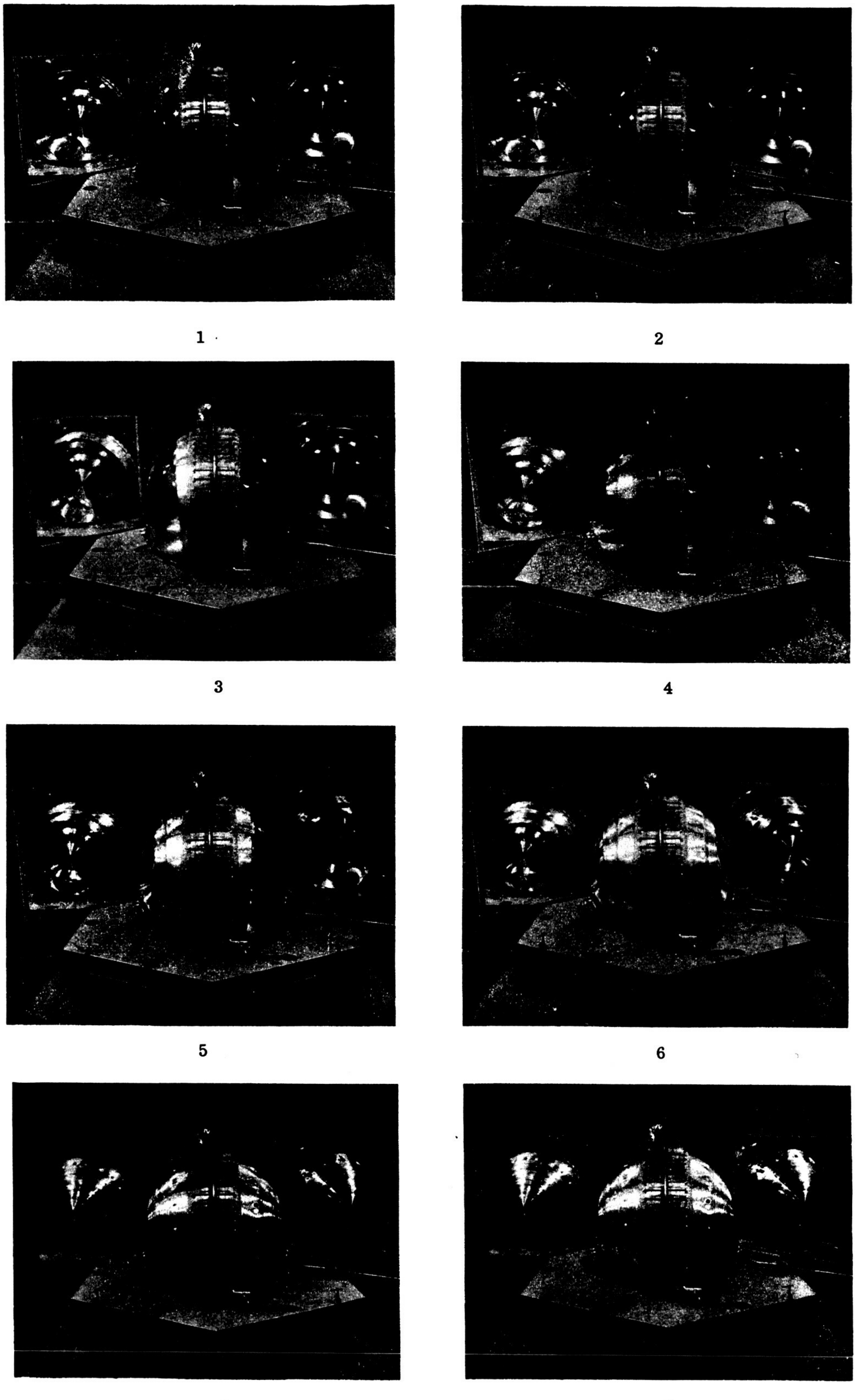


Figure 73. Convoluted Aluminum Diaphragm Expulsion Sequence

d. Surface Forces

Efforts directed to improve the reliability of positive expulsion systems by elimination of moving parts has led into investigation of surface forces of the liquid propellants to provide positioning of the propellant over the outlet port of the tank. This principle of expulsion can only be employed with propellants which wet the walls of the tanks. Hydrazine containing propellants have poor wall-wetting characteristics and will not adhere to the tank wall. Since the fuel used on the Personnel Propulsion Devices is a hydrazine containing propellant, this method of expulsion is not applicable.

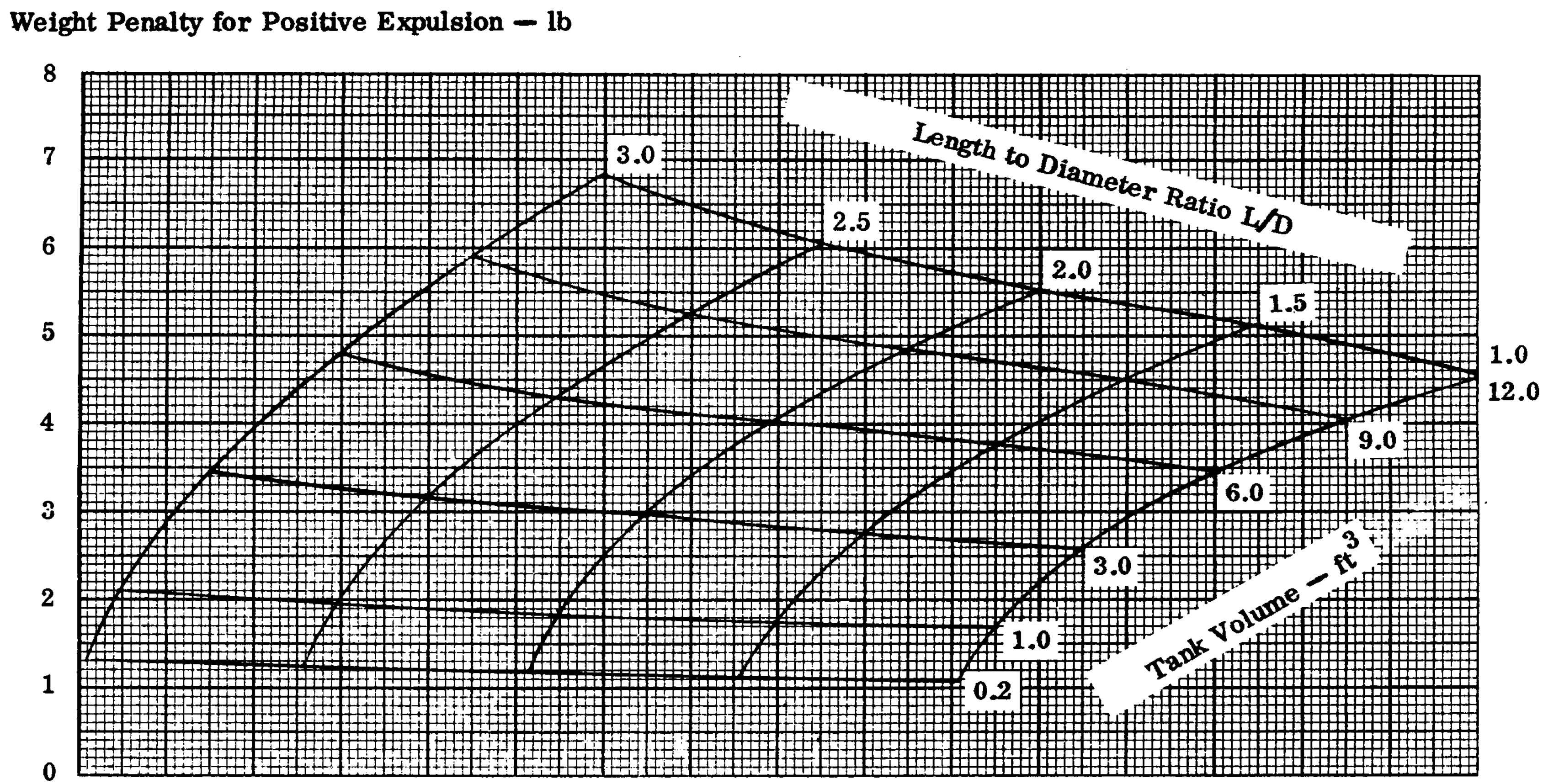
e. Spin Expulsion

Propellant orientation is possible when an angular velocity is imparted to either the individual tanks or the vehicle as a whole. Application of the latter is considered impractical to a manned vehicle. Spinning of the individual tanks introduces complexities in terms of drive mechanisms burdened by lubrication problems in vacuum, dynamic seals between the rotating tanks and the static fuel lines which present potential leaks as well as gyroscopic effects on the vehicle itself. Expulsion efficiencies attainable with a spin system are in the order of 90%.

An evaluation of the propellant expulsion techniques presented in the preceding discussion, superimposed on the mission requirements of the Personnel Propulsion Devices has yielded bladder-type tanks as the lightest, most efficient method which exhibits the greatest flexibility in regard to usage with varying tank geometry.

Based on this evaluation, parametric data has been generated to enhance the entire range of propellant tank sizes which could be used on the Personnel Propulsion Devices. The smallest tank being represented by the backpack configuration designed for 2000 fps ΔV and the largest system represented by the two man dual function vehicle capable of delivering 8000 fps ΔV. Figure 74 presents the weight penalty which must be imposed on the vehicle in order to provide positive expulsion as a function of tank volume and length to diameter ratio (L/D). The weight includes not only the weight contributed by the 9 mil teflon bladder, but also the diffuser tube; that increment of tank weight contributed by the addition of a reinforcing ring on the tank opening which is necessary for bladder insertion as well as the weight of the cover plate.

The diffuser tube diameter was also varied linearly with its length although its thickness was held constant. The change in diffuser tube diameter was necessary in order to maintain the structural integrity in buckling. The size of the tank opening has also been varied between 5.5 inches and 8.0 inches as a function of the tank diameter to facilitate bladder insertion.



Includes Weight of 3 Ply Teflon
Bladder, Diffuser, Opening
Reinforcement and Cover

Figure 74. Weight Required for Positive Expulsion

3. Tanks

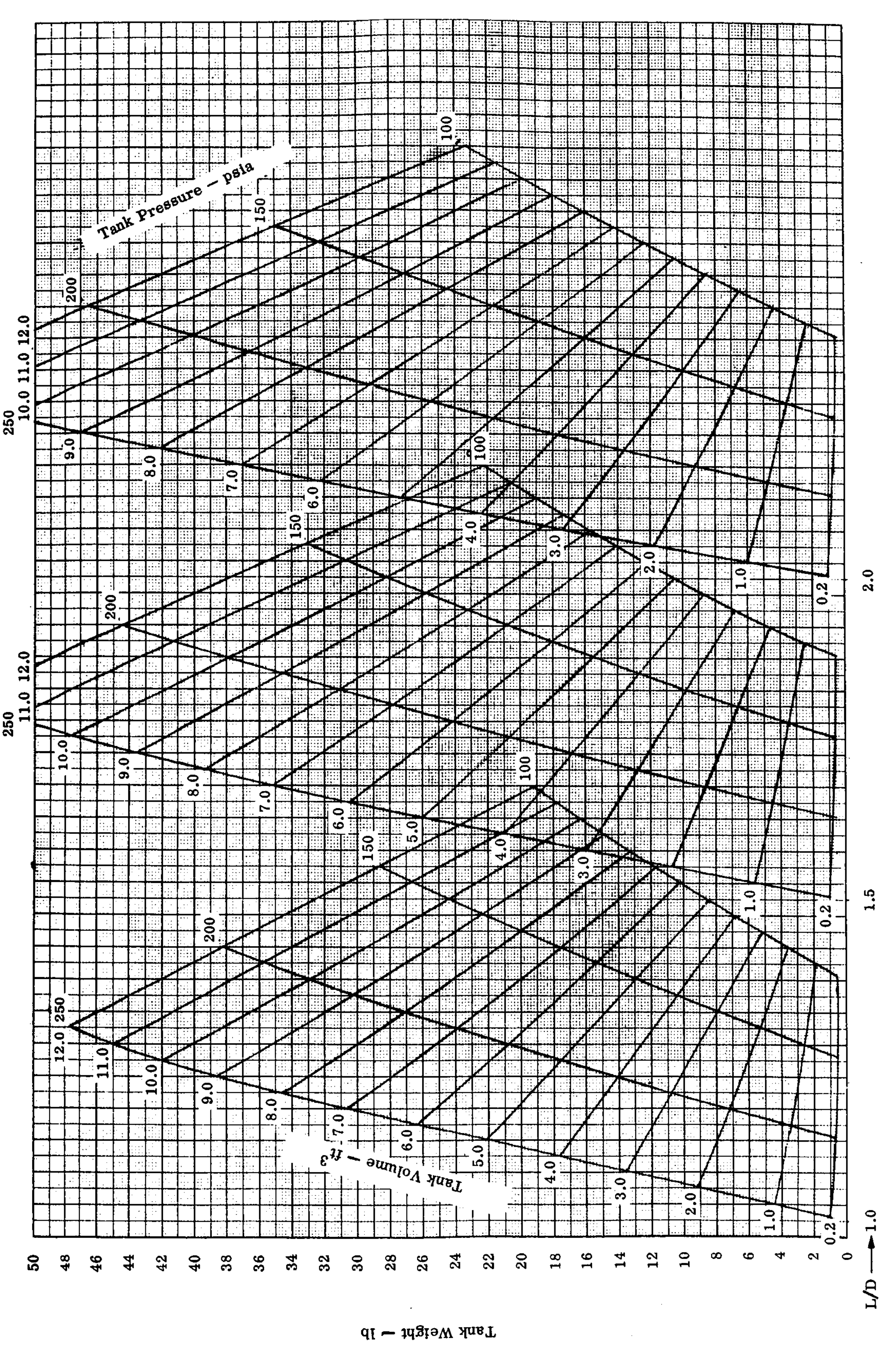
Basically, the sphere is the lightest pressure vessel shape where the design conditions involve uniform internal pressure. For the present application where the acceleration loads are small and the hydrostatic heads may be considered to be negligible because the tanks are short, the uniform internal pressure conditions is satisfied. Consequently the sphere is almost the optimum shape for this application. The sphere requires a minimum surface area to enclose a given volume; since the pressures are resisted by membrane stresses which are equal in all directions, the material is efficiently used. Against these major advantages, however, the sphere has a number of disadvantages: it does not provide for convenient attachment; and for a given volume its diameter is generally greater than that of the corresponding cylindrical tank.

In the present application where the tanks represent a small percentage of the vehicle weight, the weight penalty incurred by using the less efficient tank geometry assumes a secondary role to gross vehicle handling and dynamic considerations which may become evident during the simulation studies. For this reason, tank data have been prepared in parametric form to cover tanks of various volumes, pressures and geometry.

Based on the propulsion system optimization studies, the gravimetric mixture ratio of 1.6 established on the basis of thrust chamber operational characteristics, also yields equal volume tanks for the fuel and oxidizer. In addition, the propulsion system optimization study established the thrust chamber pressure at 80 psia for radiation-cooled chambers and 120 psia for ablative chambers. The tank design pressure is thus established by adding this pressure to the sum of the pressure drops across the regulator and lines plus the relief valve cracking tolerance. The latter results in a cumulative sum of 60 psia.

Material Considerations - Aluminum alloys, stainless steels and titanium alloys all display excellent compatibility with both nitrogen tetroxide and 50/50 fuel blend. When efficient utilization of the stress level in the material can be made, the material which displays the highest value of stress-to-weight ratio (σ / ρ) for the operational temperature range, defines the best material for tank construction. However, as the tank diameter and working pressure are lowered, other considerations become the predominating factors for minimum tank weight. The most important of these are: the minimum gage of the material required for fabrication and the fixed weight contributed by the bosses, attachment points and access holes which are invariably necessary in all tanks.

In order to minimize center of gravity excursions on the personnel propulsion devices, the propellants were divided into multiple tanks which further reduced the size of the tanks. Under these conditions, aluminum tanks were found to contribute the least weight penalty per unit volume of propellants.



Tank Pressure — psia
Tank Volume — ft³
Tank Weight — lb
L/D
1.0
1.5
2.0
50
48
46
44
42
40
38
36
34
32
30
28
26
24
22
20
18
16
14
12
10
8
6
4
2
0
100
150
200
250
12.0
11.0
10.0
9.0
8.0
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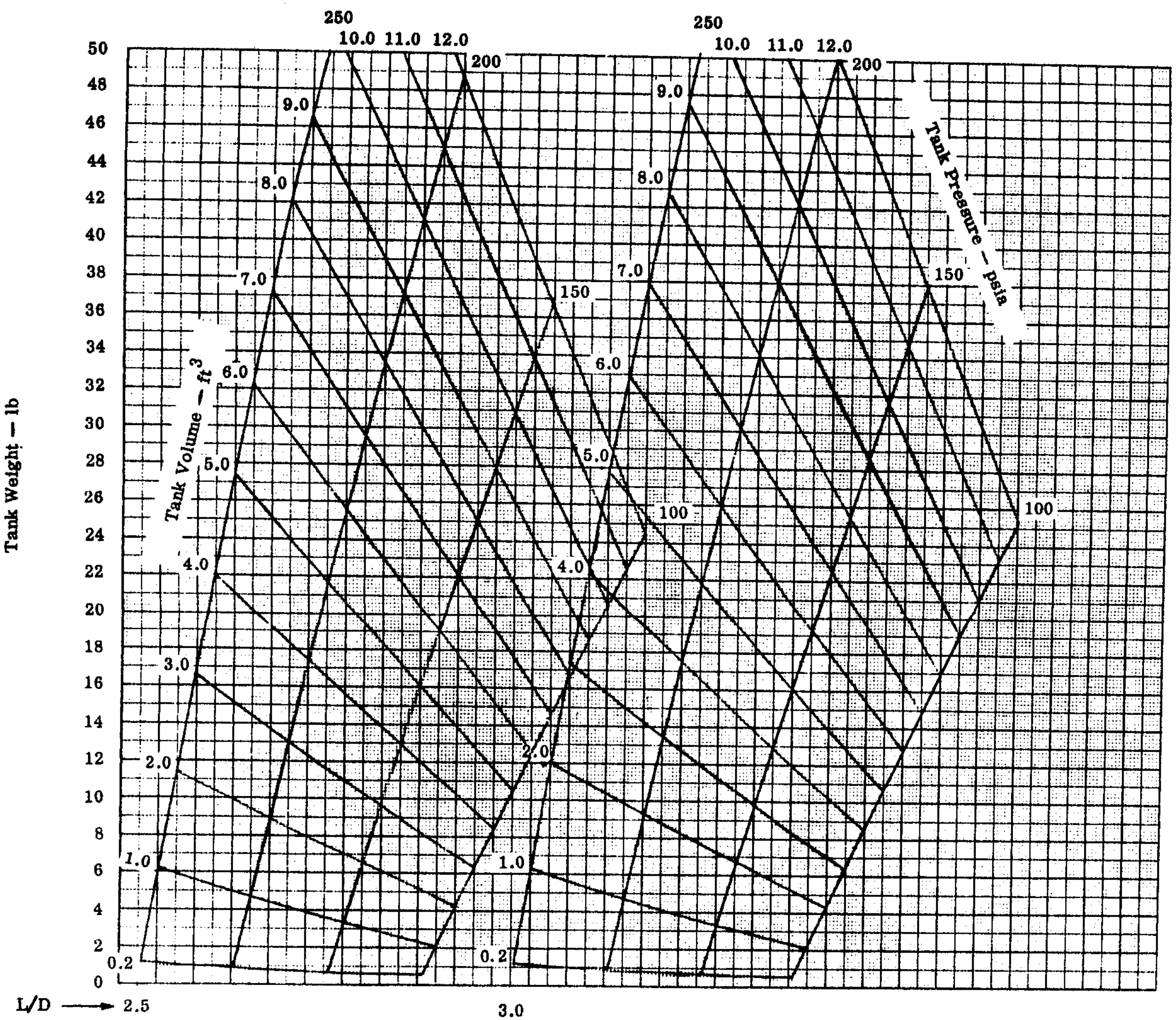
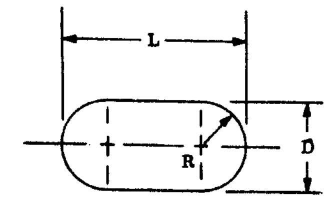
3.0
2.0
1.0
0.2





For Aluminum Alloy (6061-T6) Tanks Man Rated (FS = 2.0) With a Minimum Gage Material = 0.015. $^{+0.005}_{-0.000}$

Tank Weight Includes Knuckles and Ports.

Figure 75. Tank Weights versus Tank Volume, Tank Pressure and Length to Diameter Ratio

Based on these considerations, 6061-T6 aluminum alloy has been selected for the tank material to be used in deriving tank weights for this study - 2014 presents better parent material properties, however, it displays reduced properties in the heat affected zones of the welds. The latter necessitates the use of thicker material in the welded areas which when compensated in terms of weight for these small tanks negate the apparent advantage of 2014 aluminum alloys.

Figure 75 presents in parametric form tank weights versus volume, pressure and length to diameter ratio for tanks constructed from 6061-T6 aluminum alloy employing man rated factors of safety (see Section VI). A minimum gauge of 0.015 inches has been established as minimum shell thickness based on fabrication considerations with a scratch allowance of 0.005 inches.

Spherical tanks are constructed of two hemispheres welded at the equator. Hemispherical ends are also employed on all cylindrical tanks with weight allowances made for the knuckles (the transition from hemispherical end to cylindrical section of the tank). Adequate reinforcement has also been provided in the calculations to enable mounting of the tanks on the vehicle structure. The weights depicted in Figure 75 do not include the weight penalty imposed when positive expulsion is desired; however, it may be synthesized by adding corresponding weights given in Figures 74 and 75.

The length and diameter dimension of the tanks are shown in Figures 76 and 77 respectively as a function of tank volume and (L/D).

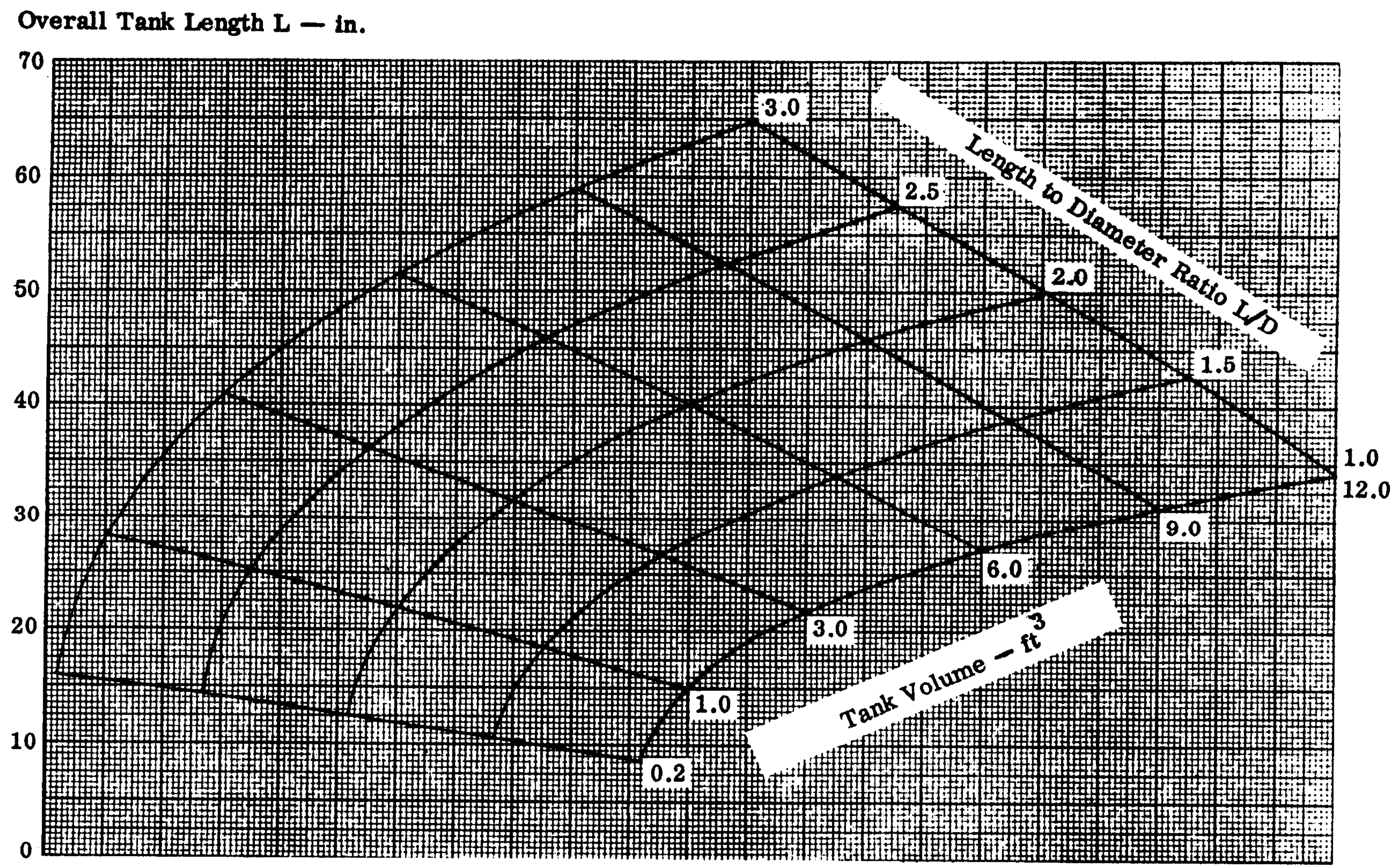


Figure 76. Length and Diameter Dimension as a Function of Tank Volume

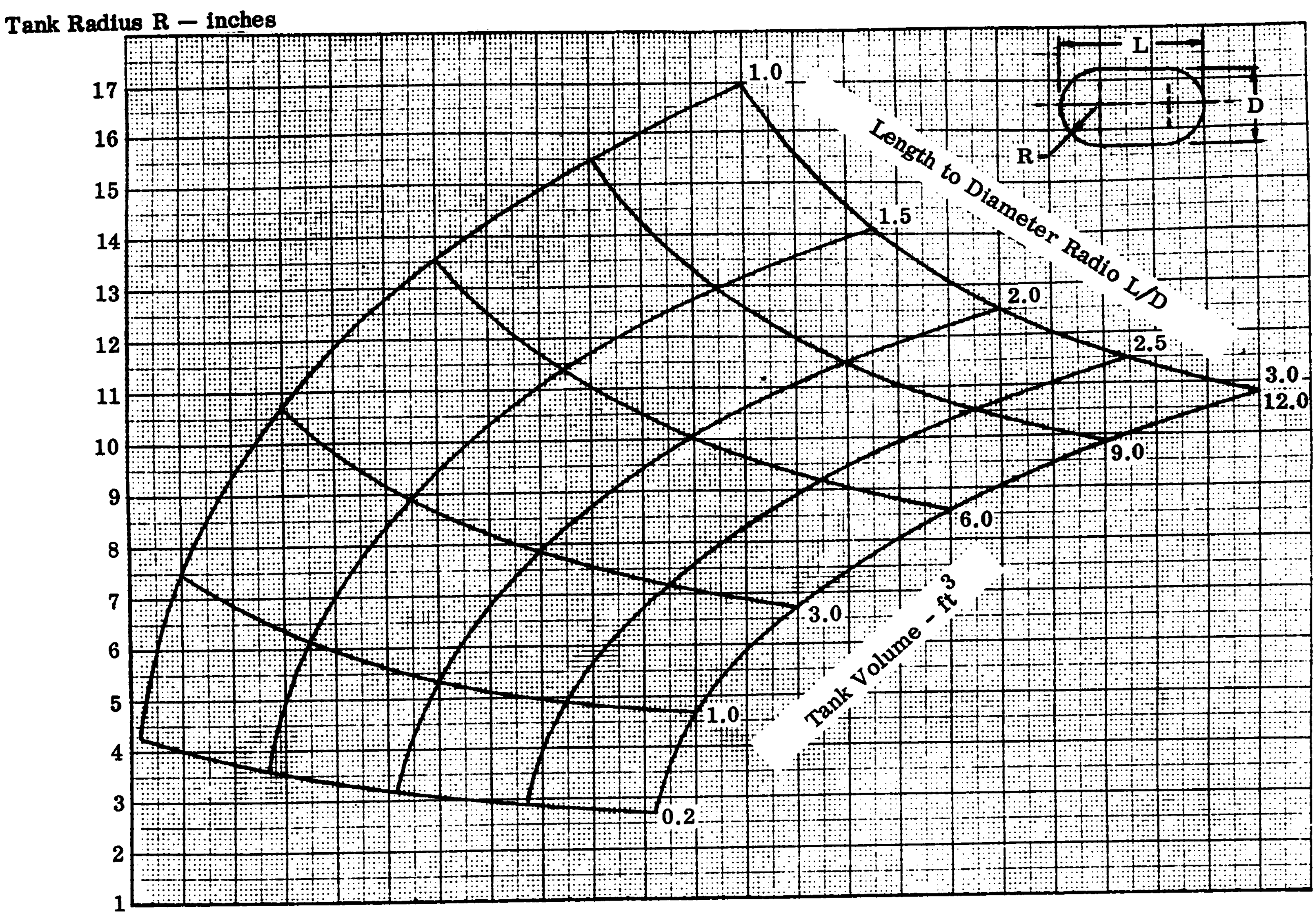


Figure 77. Length and Diameter Dimensions as a Function of L/D

V. ELECTRONIC - ELECTRICAL EQUIPMENT

A. APPROACH

The nominal ΔV requirements imposed for transportation, escape and dual function personnel propulsion devices have been estimated on the basis of making maximum utilization of the man's sensory and control abilities and, therefore, using a minimum amount of vehicle equipment such as sensors, displays and auxiliary flight control, navigation and guidance equipment. The accuracy however, that can be attained in orbital rendezvous (or the target miss diatance on surface translations) can only be estimated by extensive simulation studies.

In order to be able to determine, through simulation studies, the tradeoffs which exist between equipment sophistication and propellant consumption, four potential groups of electrical and electronic equipment designated as groups A, B, C and D, which are described later in the text, have been developed. These tradeoff studies should be conducted by increasing the vehicle sophistication through addition of the electronic equipment, displays and navigation aids which would tend to improve the rendezvous accuracy, enable the astronaut to approximate a flight path which approaches the optimum for the specific mission and determining the effect on propellant requirements.

The design philosophy employed on the vehicles presented in Section VII, which are submitted for final configuration selection through simulation studies, was such that these configurations could be subjected to study with the characteristics of each of the equipment groups based on the following recommended procedures:

(1) Initiate vehicle evaluation with a design which incorporates the minimum of electronic equipment and displays such as incorporated in Group A.

(2) Fly this vehicle on the simulator with these characteristics and monitor the performance parameters for its intended mission (controllability, response, energy expenditure, ΔV cutoff accuracy, etc.).

(3) From the simulation runs, identify the areas in which additional sensors and/or electronic equipment could produce a marked improvement in vehicle performance.

(4) Substitute on the vehicle the more complex group of equipment (B, C or D) or select from the furnished table, the equipment which provide the desired functional characteristics and impose on the vehicle the weight penalty increment which these equipment represent. An addition or increase in the power supply, by an increment necessary to accommodate the additional equipment, must be made.

This procedure may be repeated until the desired vehicle characteristics have been established. In view of the wide scope of missions to be simulated, the additional equipment required would vary with each mission profile.

B. ELECTRONIC ELECTRICAL SYSTEM DEFINITIONS

In order to provide mission flexibility, information has been accumulated for present and advanced systems to provide communications, telemetry, sensors, displays and navigational equipment together with thin physical characteristics (weight, volume and power consumption) and their functional characteristics (dynamic range of coverage, accuracy/resolution, etc.).

The vehicle guidance, navigation and attitude systems are required to determine more accurately the relative position and terminal velocity of the vehicle during flight. This requirement arises from uncertainties in burning time, vector orientation accuracy, as well as the cumulative contribution of the attitude stabilization system when operated in the unbalanced mode (not pure couples). In order to provide realistic mission profiles, the expected accuracies of these systems have been presented in Table VI for equipment which can be installed on the vehicle.

The equipment presented in Table VI may be either employed individually or in groups to correct any deficiencies which may arise during the simulation studies. Four suggested groupings may be considered which comprise systems of increasing order of complexity these are listed below:

Group A - consisting of electrical and electronic equipment whose function is mainly to provide inputs to the pilot in terms of displays which are essential for vehicle control. Guidance equipment is limited to a simple sighting device.

Group B - provides an improved velocity sensing system over that employed in Group A and an optical sight system which improves the sighting accuracy.

Group C - provides an attitude reference system with an artificial horizon indicator and displays attitude, altitude and altitude rate.

Group D - provides radio-radar type equipment similar to those used on the LEM vehicle. The sensor outputs from this equipment can either be displayed to the pilot or fed to an airborne computer.

The specific equipment included in each group, the component weights, power requirements, and a system description for each group are discussed in the following paragraphs.

TABLE VI

PERSONNEL PROPULSION DEVICES, ELECTRICAL EQUIPMENT LISTING AND CHARACTERISTICS

No.	Used in Groups	Equipment Description	Typical Manufacturers	Size & Weight	Primary Power Watts	Functions and/or Measured Quantities	Accuracy Resolution or Figure of Merit	Dynamic Range or Coverage	Features or Advantages	Comments
1	ABCD	Checkout System, Meter and Lighted Display, Limit Detectors & Comparators	Stromberg Div. Gen. Dynamics Bell Aero. Systems	30 cu. in. 2 lb	3.0	Go-No-Go Checkout	As Required By Individual Function to be Checked	All Electrically Sensed Functions	Provides Pre Flight Confidence Check	
2	ABCD	Power Source, Nicad or Silver Cell Battery	Gould National, Union Carbide, Yardney, General Electric Co.	Variable Depending on Load	Approx. 40 Watt hours/lb	28 Volt D.C. Primary Power Supply	+30 Volt at Full Charge +24 Volt Near Exhaustion	Sufficient Capacity to Handle all Loads for 20 Minutes	50 Percent Reserve Capacity	Rechargeable
3	ABCD	c.g. Sensing, Strain Gauges on Landing Gear, Meter Readout	Weston Instrument Co. Div. Daystrom Corp.	10 cu. in. 1.0 lb	0.1	Sense Center of Gravity	2 Percent of Full Scale	Maximum Range of Stabilization Control Capability	Allows Manual Alignment of Vehicle c.g. with Thrust Vector	Minimizes Reaction Jet Thrust Fuel Use, Provides Safety Factor
4	BCD	Communication, 2 Way CW Radio, S Band Earth-LEM-Led-Apollo Cmd. Mod.	Collins Radio Co. Radio Corp. of America, Motorola	1000 cu. in 35 lb	106 Peak 22 Avg.	Voice Communication	Crystal Controlled Phase or FM Modulation	240,000 Miles to Earth	Compatible with DSIF Provides Transponder Ranging & Voice	Assume 20% Duty Cycle for High Power X Mitter During 20 Minute Mission
5	ABCD	Control and Display Panel Console, Switches, Meters, Hand Controls	Bell Aerosystems Panel Weston Meters	510 cu. in 12 lb	2.0	Mounting for All Controls & Displays	As Required for Individual Functions	All Control or Display Functions	Central Control and Display Panel-Console	Common Lighting Functional Layout
6	ABCD	Clock Timer, Setable, Interval and Absolute Timing	Cramer Div. Gianini A.W. Hayden	30 cu. in. 2 lb	1.0	Time, Time Intervals	0.01 Sec Switching 0.10 Sec Readout	24 Hours Clock 100 Minute Timer	Pre Set Interval Timing Electrical Switching Action	Visual Readout and Switch Closures
7	ABCD	Stabilization & Main Lift Rocket Engine Control Circuits	Bell Aero., Circuits Fairchild, Texas Instru. Semiconductors	130 cu. in 4 lb	12.0 Avg.	Energize Rocket Control Valves, Provide Logic	Valves Full on or Full Off	Controls all Vavle Cicuits	Provides Engine Out Logic, Selected Group Firing, Attitude Control Selection	Switches Heavy Valve Current Loads
8	A	Open Frame, Simple Mirror Type Optical Sight	Bell Aerosystems	100 cu. in 4 lb	None	Angular Sight Reference	1.0 Minute Static 1.0 Degree Dynamic	180 deg. Pitch Fixed Roll & Yaw	90 Degree Field of View in Azimuth & Elevation	Limited By Operators Pressure Suit Face Plate and Vehicle Structure
9	BCD	Simple Telescope Sight, Tiltable in Pitch	Unitron, Bausch & Lomb American Optical	80 cu. in 15 lb	None	Angular Sight Reference	1.0 Minute Static 0.2 Degree Dynamic	180 deg. Pitch Fixed Roll & Yaw	4 Power, 14 deg. Field of View	Limited by Vehicle Structure & Operator
10	BCD	Accelerometer & Integrator	Kistler Instruments, Bell Aero. Systems, Kearfott Div. GPL	20 cu. in. 2.0 lb	2.0	Acceleration Integral, Velocity Measurement	10^{-5} g Error From All Sources	10^{-6} g Threshold 10 g Full Scale	Measures Acceleration Along Major Thrust Axis	Does Not Sense Lunar Gravity Effects, But These Can Be Resolved
11	CD	Rate Gyro and Integrator Assembly, 3 Axis	Fairchild Camera & Inst. Sanders Assoc.	260 cu. in 4.0 lb	8.0	Attitude Reference Flight Control	0.1 Degree Drift in 10 Min.	±10 Degree in Roll & Yaw ±50 deg Pitch	Fixed Miniature Gyros Solid State, Chopper Stabilized Integrators	Relatively High Drift Rates, Can Be Updated By Optical Reference
12	C	Radar Altimeter	Raytheon Co. Laboratory For Electronics	300 cu. in 10 lb	15.0 Avg.	Altitude & Altitude Rate	2 Ft Up to 200 ft 1% of Scale Above	100,000 Feet Max. 10 Feet Min.	Measures Altitude Above Lunar Terrain	Fixed Antenna, May Require Pitch Gimbal
13	D	Landing Radar, FM/CW Modulation, Doppler Rate Sensing	Ryan Aeronautical	3400 cu. in. 30 lb	130	Altitude, Alt Rate Horizontal Forward & Side Velocities	1/4% of Scale Rates 1% of Altitude	100,000 Feet Alt. 6000 fps Alt Rate 600 fps Horizontal Rate	Fixed 4 Beam Antenna (2 Pitch Positions) Used on LEM	Provides Landing Data For Lem in Conjunction with Computer
14	D	Rendezvous Radar, X Band Gimballed Antenna	Radio Corp. of America	1920 cu. in 60 lb (Not. Incl. Antenna Volume)	150 Estimated	Range, Range Rate Az. & Elev. Angle	1/4% of Range Rate 1% of Range	400 Mile Range 6000 fps Rate	Operates with Transponder Used in Apollo and LEM	Used in Conjunction with Raytheon Computer

1. Group A Equipment

Item No.	Description		Power Watts	Weight Lb
(1)	Go-no-go checkout system		3.0	2.0
(2)	Power Source (battery)	Equipment	-	1.0
		Prop. System	304.0	15.2
(3)	c.g. sensor		0.1	1.0
(4)	Communications - LOS		22.0	4.0
(5)	Controls and Displays, simplified panel		2.0	8.0
(6)	Clock timer		1.0	2.0
(7)	Stabilization and control circuit		12.0	4.0
(8)	Optical sight, open frame type		-	4.0
		Total	344.1	41.2

In this group, as in all groups, a certain minimum of equipment is assumed to provide such functions as preflight checkout, equipment power source, center of gravity location sensing and communications. The power source is assumed to be a battery and for this simple system (Group A), it will be somewhat smaller than the more complex systems.

The c.g. sensor system consists of strain gauges mounted on the landing gear legs so that after pilot and payload are aboard the vehicle, the c.g. can be found and if off center, the c.g. can be shifted by manually moving the pilots seat prior to launch. This keeps the c.g. within the capability of the stabilization reaction control jets and minimizes their thrust level and fuel usage.

The communications system is assumed to be compatible with that on the Apollo command module and that in the LEM. It is operable in line-of-sight applications, but does not have the power capability to return a signal directly to earth or to receive a signal from earth. It consists of an all solid state UHF or S band equipment having two-way (simplex) voice capability. A fixed, relatively broad coverage upward facing antenna is used, which can be redirected manually for horizontal communications to LEM or for communications with the command module.

The control and display console includes the hand controller for attitude control. It provides the command signals to the reaction control stabilization system. Also included in the control section are the main lift rocket engine firing controls. The displays for the simple system (Group A) consist mainly of the clock timer, the c.g. sensing and checkout display.

The clock timer is provided for the purposes of timing the start of a flight involving a rendezvous and for subsequently timing the rocket engine firing duration. If the flight is of the transportation type, the timer is used for timing the rocket firings program including retroaction in preparation for landing. A clock is also required for celestial navigation purposes if such navigation techniques are to be used.

The circuits required to drive the stabilization reaction jets and the main lift rockets in an acceleration command mode are of all solid state construction and provide the rocket valve driving power. The components in the stabilization and control circuit include the throttle logic, throttle and on-off valve driver, and the reaction control valve driver for (6) thrusters. The engine controls and timer will be the same for all groups except that in the more complex systems, signal inputs and outputs will be connected for partially automatic operation. In the simpler systems, all operations are assumed to be manual.

The optical sight is an open frame type using a grid line reticule. No automatic stabilization is provided here. The sight is used by setting the desired elevation angle by dial. The reticule is then held on the selected sighting object (earth, star, command module, lunar landmark, etc.) by maneuvering the vehicle by use of the reaction control stabilization jets. The accuracy of such a simple sight would probably be in the order of one minute of angle under static conditions but considering vehicle motion, vibration and human factors, the expected accuracy would be only about one degree.

In all systems, a 20-minute maximum flight operating time is assumed for purposes of battery watt hour rating and weight estimation. Some battery loads are statistically inactive for portions of the flight time. For example, the communications transmitter draws about 106 watts during the time it operates but it is assumed that it operates only about 20 percent of any flight operation. Therefore, the communication set is shown as a 22 watt load.

2. Group B Equipment

Item No.	Description		Power Watts	Weight Lb
(1)	Go-no-go checkout system		3.0	2.0
(2)	Power source	Equipment	-	1.2
		Propulsion	304.0	15.2
(3)	c.g. sensor		0.1	1.0
(4)	Communications		22.0	35.0
(5)	Controls and Displays		2.0	12.0
(6)	Clock timer		1.0	2.0
(7)	Stabilization and control circuits		12.0	4.0

Item No.	Description	Power Watts	Weight Lb
(9)	Optical sight, simple telescope type	-	15.0
(10)	Accelerometer and integrator, single axis	2.0	2.0
	Total	346.0	89.4

In this group, the equipment is much the same as in Group A except for items 4, 9 and 10. The communications system in this and subsequent groups is compatible with the Apollo command module and the LEM. Unlike that included in Group A, however, it does have the power capability to transmit directly to earth and receive signals from earth, thereby providing communication capability beyond line-of-sight applications. The communication set shown is the Collins S band high power unit used with the DSIF (Deep Space Instrumentation Facility) aboard the Apollo command module. It may be possible to find a smaller, lighter weight communication set particularly if communications to earth are not required. The LEM and Apollo command module will also have VHF communications equipment and a set of compatible VHF radio equipment should be lighter and require less power. The optical sight is of the simple telescope type and although it is not stabilized, it affords some magnification (about four power, 14 degree field of view) and should therefore improve the sighting accuracy to about 0.1 degree compared to 1 degree attainable from the open frame sight.

An accelerometer and integrator system is added to improve the velocity sensing system which had been done by timing only in system A. The accelerometer is used to measure acceleration along the major thrust axis. A further improvement in this system would be to allow the accelerometer-integrator system to cut off the main rocket lift engine at a preset ΔV thus eliminating the human reaction time error. However, since the intent is to check that capability of the human pilot, a meter readout of the accelerometer is provided. The pilot then manually cuts off the rocket engine when the desired value of velocity has been achieved.

The weight of the telescopic sight is mainly attributed to the heavy optics required for eye relief when operated with a pressure suit face plate. Periscopic optics may also be required to get the line-of-sight around the vehicle structure.

3. Group C Equipment

Item No.	Description		Power Watts	Weight Lb
(1)	Go-no-go checkout		3.0	2.0
(2)	Power source	Equipment	-	2.5
		Propulsion	304.0	15.2
(3)	c.g. sensor		0.1	1.0
(4)	Communications		22.0	35.0
(5)	Controls and Displays		2.0	12.0
(6)	Clock timer		1.0	2.0
(7)	Stabilization and control circuits		12.0	4.0
(9)	Optical sight, simple telescope type		-	15.0
(10)	Accelerometer and integrator, single axis		2.0	2.0
(11)	Rate gyro and integrator system, 3 axis		8.0	4.0
(12)	Radar altimeter		15.0	10.0
		Total	369.1	104.7

In this group, a rate gyro and integrator system have been added to provide a rate command system and a small, light weight attitude reference system which will provide reasonable attitude information over the short time of flight. The attitude information is displayed on a conventional artificial horizon type indicator. The attitude information after 15 minutes is expected to be in error by between 0.1 and 1.0 degree. The attitude system may be corrected in flight by the operator if sufficiently good optical reference data can be obtained.

A radar altimeter has also been added to this system and should be of considerable value if the flight is of the lunar point to point type. It will provide altitude and altitude rate information and will be most useful in the retro firing operation preparatory to landing. However, by using the altitude rate feature during the boost phase, a measure of the boost velocity is made and can be used to check or correct the reading obtained by the accelerometer-integrator.

If the flight is to be of the orbital rendezvous type, the radar altimeter might be replaced with a rendezvous radar transponder. This piece of equipment is built by RCA and is used for the LEM-Apollo command module rendezvous. It is of approximately the same weight and requires approximately the same power as the radar altimeter. In operation with the transponder, the rendezvous radar aboard the command module would locate the approaching vehicle and could either maneuver the command module to the lunar escape device or could send voice instructions for maneuvering the LED vehicle to the command module.

4. Group D Equipments

Item No.	Description		Power Watts	Weight Lb
(1)	Go-no-go checkout		3.0	2.0
(2)	Power source	Equipment	-	12.5
		Propulsion	304.0	15.2
(3)	c.g. sensor		0.1	1.0
(4)	Communications		22.0	35.0
(5)	Controls and Displays, extended panel		4.0	16.0
(6)	Clock timer		1.0	2.0
(7)	Stabilization and control circuits		12.0	4.0
(9)	Optical sight, simple telescope type		-	15.0
(10)	Accelerometer and integrator, single axis		2.0	2.0
(11)	Rate gyro and integrator system, 3 axis		8.0	4.0
(13)	Doppler radar (similar to LEM landing radar)		130.0	30.0
(14)	Rendezvous radar (similar to LEM rendezvous)		150.0	60.0
	Total		636.1	198.7

In this group, a complete set of radio-radar type sensing equipment has been added so that the vehicle performance at any time in either the lunar point to point flight or in the orbital rendezvous flight, can be measured by external sensing. The sensor equipment in this group is almost equivalent to that being used on the LEM vehicle. However, because the object is to investigate the human capability rather than automatic equipment, the sensor outputs will be displayed rather than fed to a computer.

If it is desired to automate the system, the LEM computer built by Raytheon is available. Other smaller and simpler computers are also available in both analog and digital form.

The landing radar provides altitude, altitude rate, forward horizontal velocity and cross track horizontal velocity relative to the lunar surface. The rendezvous radar provides range, range rate, and two orthogonal angles relative to a transponder equipped point target (the command module or LEM). The rendezvous radar has some limited range noncooperative capability whereby it would function as a backup for the landing radar or would work (at reduced range) without a transponder on the skin echo (radar signal bounce) off the target spacecraft (Apollo command module).

VI. DESIGN CRITERIA

A. GENERAL

This section presents a summary of the design criteria which have been investigated during the course of the study in order to identify problem areas and conditions which dictate design trends, and to establish acceptable solutions which result in realistic vehicle weight and performance estimates. The design criteria have been grouped into three major areas:

(1) Structural Criteria: Identification of handling, transportation and operational loads, definition of proof pressures and factors of safety to be used in the vehicle design phases.

(2) Thermal Control: Identification of the mean bulk temperature and the amplitude variation of the propellants stored within the vehicle tanks, and definition of the insulation requirements for the most severe design conditions represented by the 180 day storage period on the surface of the moon.

(3) Micrometeorite Protection: Definition of impact and penetration probability and protection requirements for the vehicle during the 180 day storage period and for the man during operation of the vehicle.

B. STRUCTURE

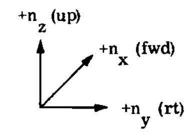
1. Structural Design Criteria

The structural design of a vehicle is influenced by many interacting factors some of which are very significant and others of which have little effect on overall vehicle configuration. In the present study, where the main interest is in gross vehicle configuration, the structural design is actually an input of relatively low importance to the main design area. Since the main area of interest is guidance, control and propulsion of vehicles with different performance capability, the structural design and analysis completed in this contract has been taken to the point where the weight and inertia characteristics, as related to control performance requirements, are defined and the general feasibility of mounting the hardware required for the various types of guidance propulsion, and control systems is established; thereby establishing a typical feasible structure for each vehicle.

The approach used was to establish structural design criteria, then based on these criteria to estimate sizes of the major airframe members of each vehicle, and finally to perform parametric studies of one type of landing gear. The structural design criteria are summarized in Table VII. Presented in this table are the loads to which the personnel propulsion devices will be subjected during their mission life. These include earth transportation, delivery to the moon and operational flight loads. All items listed are not critical for the design of all the vehicles but are shown to demonstrate this fact. For instance transportation on earth and booster acceleration

TABLE VII

PERSONNEL PROPULSION DEVICES - STRUCTURAL DESIGN CRITERIA ⚠2



1. TRANSPORTATION ON EARTH (AIR TRANS)

Direction ⚠1	Ultimate Load Factors (Uncombined)	
Longitudinal	+1.5, -4.0	(n_x)
Lateral	+1.5	(n_y)
Vertical	+4.5, -2.0	(n_z)

2. BOOSTER ACCELERATION

Direction ⚠1	Limit Load Factors (Uncombined)	
Longitudinal	6.0	(n_x)
Lateral	2.0	(n_y)
Vertical	2.0	(n_z)

3. LEM TRUCK LANDING/DOCKING ACCELERATIONS

Direction ⚠1	Limit Load Factors (Uncombined)	
Longitudinal	4.0	(n_x)
Lateral	4.0	(n_y)
Vertical	8.0 within 30° cone angle of z axis	

4. LUNAR ENGINE OPERATION

Direction	Limit Load Factors ⚠3 (Uncombined)		
Longitudinal			
Lateral			
Vertical	1.5	(n_z)	Burnout
	0.5	(n_z)	Takeoff

Thrust Overshoot Factor = 1.25 Unthrottleable Engines
Thrust Overshoot Factor = 1.00 Throttleable Engines

5. LUNAR LANDING (TRANSPORT OPERATION ONLY)

Mode	Limit Sink Speed (FFS)	Ultimate Sink Speed (FPS)
Level Terrain	$V_v = 20.0$	$V_v = 30$
Level Landing Approach		
20° sloped terrain - Level Approach		
Combined vertical and side drift landing		

6. PRESSURE VESSELS

Tank	Usage	F_y ⚠4	F_u ⚠4
Helium Storage	Non-hazardous	1.20	1.60
	Hazardous	1.50	2.00
Propellant	Non-hazardous	1.0	1.25
	Hazardous	1.50	2.00

7. STRUCTURAL FACTORS OF SAFETY

Usage	F_y ⚠5	F_u ⚠5
Non-hazardous	1.0	1.25
Hazardous	1.0	1.50

8. TEMPERATURE EXTREMES

-240° F ⟶ 240° F

NOTES:
⚠1 Relative to transport vehicle
⚠2 Load factors based upon earth G
⚠3 Escape vehicle factors to be specified for each design
⚠4 Factor to be applied to working pressure per MIL-T-5208A
⚠5 Factor to be applied to limit load

load factors are not as severe as LEM truck landing/docking load factors. The transportation on earth load factors are the aerial delivery restraint factors of MIL-A-8865(ASG). Booster acceleration and LEM truck landing/docking accelerations are the current maximum design load factors for the Saturn 5 booster and the truck version of LEM respectively which have been assumed to comprise the vehicle which will deliver the personnel propulsion devices to the moon. Lunar engine operation load factors reflect the proposed range of thrust to weight ratios for the family of lunar flying vehicles.

The limit sink speeds for lunar landing presented in Table VII are based on current LEM requirements for landing on a slope, with superimposed bumps and depressions, at touchdown velocities of 10 feet per second vertical and 5 feet per second horizontal. A landing of this nature imposes maximum loads on one leg of the vehicle only so that the capability of all the legs combined must be greater than that actually required for any one landing. This leads to an equivalent capability of 20 feet per second vertical velocity when landing on all four legs on level ground with no lateral velocity. This is a more straightforward condition to analyze and it requires the same gear capability as the (10, 5) fps landing on a 15° slope, so it is logical to use this symmetrical condition in a study of this nature to avoid lengthy analysis. It should be emphasized that the 20 feet per second vertical velocity condition is a design condition only and it should not be implied that the vehicle could be landed with this large vertical velocity since that would leave no capability to accommodate lateral velocities and/or slopes. A capability of 1.0 V feet per second on four legs implies an actual landing capability of approximately 0.5 V feet per second in the vertical direction combined with 0.25 V feet per second in the horizontal direction on a 15° slope. The pilot would be restricted to operation within this latter envelope.

Design ultimate landing speeds are shown to be 1.5 times limit speeds. This reflects the capability of a spring type landing gear with an ultimate strength of 1.5 times limit strength. Since the deflection of this type of gear at ultimate load is 1.5 times the deflection at limit load, the energy capability at ultimate is $(1.5)^2$ times the capability at limit, while the velocity capability at ultimate is thus 1.5 times that at limit. Other types of gears which are velocity sensitive may have an ultimate load 1.5 times limit load but no greater deflection, so that ultimate velocities are 1.5 times limit velocities. Still another type of gear with a constant load characteristic may not have any capability above limit velocities.

Pressure vessel design factors shown in Table VII are typical of requirements for manned space vehicles. The structural factors of safety are standard for vehicles of this type. Nonhazardous structure is that which would not bring injury to the crew if it failed. In a vehicle where even a navigation failure could be termed catastrophic due to landing too far from oxygen resupply, it is apparent that not only the primary airframe structure but also any secondary structure necessary for the completion of the mission should be designed to the hazardous factors.

Estimates of the airframe member sizes involves first a definition or selection of the airframe arrangement followed by calculations as to the strength required. Definition of the airframe arrangement is closely interrelated to configuration selection. Once the configuration is fixed the choice of variations in the structural arrangement is usually very limited. In general the configuration of the vehicle dictates the geometry of the airframe, and the smallest total weight of structure will occur with the configuration having the least amount of structure. Configuration selection often becomes a search for the configuration having the smallest airframe. For the vehicles in this study where specific propulsion engine and reaction control combinations with various fuel tank arrangements are the main areas of interest, it is not difficult to define the airframe arrangement required to support the engines, tanks, and crew. The type of landing gear also influences the airframe layout in regards to provision of mounting points but this has not been made a variable in this study as only one type of landing gear has been used on all vehicles. Although the landing gear loads may be the most severe forces applied to the portions of the structure to which the landing gear is attached, different types of landing gear will not necessarily result in a significant variation in total airframe weight since the ground reactions are the same for all the gears, as explained in a following discussion.

Truss structure is used on all the vehicles presented in Section VII since it is ideal for vehicles not requiring aerodynamic fairings and must be compatible with multiple concentrated loads. In many cases, built-up sheet metal members with shear webs and beam caps would be equally suitable. However, the difference in weight between a truss and a sheet metal structure is insignificant, compared to the total weight of each vehicle, so that the truss structure shown is representative of what is required and is therefore typical. Aluminum alloy has been used in all the airframe structures during this study. Although steel and titanium do have higher strength to weight ratios this is not usually an advantage in normal airframe structure except in highly loaded fittings where size is limited. Most airframe structure is critical in compression or bending and in these cases the low density of aluminum permits thicker gages and higher buckling allowables with less weight. Theoretically magnesium is better than aluminum but the percentage gain is very small and the weight of each airframe can be considered typical for a magnesium structure also. Beryllium would be ideal since it has higher buckling allowables and lower density than aluminum but it is not state-of-the-art for airframe structure of this nature.

2. Landing Gear

The landing gear used on the transportation devices must satisfy the following objectives:

(1) Provide multiple landing capabilities with little or no maintenance.

(2) Limit the landing load factor magnitude and duration to tolerances acceptable for manned operation.

(3) Represent the least weight configuration for the intended mission.

Several approaches are possible toward the design of a landing gear to meet these objectives. The three types considered for use on the Personnel Propulsion Devices are:

Energy absorption gears using crushable honeycomb for energy dissipation
Viscous damped systems (oleo strut) and
Spring gears.

Energy absorption gears have been subjected to extensive studies by Bendix (Reference 1). These gears provide good stability characteristics both in level and inclined planes and represent a relatively lightweight arrangement, second only to the spring gear. However, gears using crushable honeycomb do not meet the objectives set forth herein. Present designs do not provide multiple landing capability and require cartridge replacement after a relatively small number of landings. The gear investigated for comparison was comprised of four pads with a tripodal arrangement to support each pad. In order to provide vertical and lateral energy absorption capability a crushable honeycomb cartridge is required on each of the struts converging to the pad. After each landing, all cartridges must be inspected and possibly replaced before initiating the return trip.

Viscous damped systems can not be applied to lunar applications in their presently developed condition. Problems arise from the employment of dynamic seals for operation in vacuum, and temperature conditioning is required to control the viscosity of the working fluids within the oleo. Whereas temperature control in propellant tanks reduces to effectively controlling the radiation interchange between the propellant tank and the surroundings, heat transfer to the fluids in the oleo is mainly by conduction from the lunar surface and the strut structure.

Spring gears offer several advantages: they are simple, lightweight, have no moving parts, and are completely reusable. Against these advantages, however, they pose one main disadvantage. They provide little or no internal damping, thus imposing landing velocity and plane inclination limitations. Because of the ability of this gear to meet the initial objectives, subsequent effort was directed toward development of characteristics and definition of the limitations of the spring gear.

If it is desirable in the simulation studies to utilize landing gears other than the spring type gear configuration data presented in Section VII must be modified, although as a consequence of this change the difference may not be significant. For instance, a crushable honeycomb gear with twelve telescoping struts arranged into four tripods would weigh 30 pounds more than the spring gear for a 1400 pound vehicle (including 2 man crew). An empirical correction for the moment of inertia would be to add one pound for each 200 pounds of vehicle weight at a position midway between each landing pad and the landing gear attachment fitting on the body. This does not indicate a major correction to the vehicle moment of inertia. As a consequence of these facts it is concluded that the airframe data presented in this report are representative for vehicles of classes studied and that the designs employing the spring type gears are valid representations for simulation purposes of vehicles employing other landing gear designs.

The landing gear parametric studies which were performed for the cantilever spring type landing gears employed Fiberglas reinforced plastic (with ultimate allowable stress of 67500 psi and an E of 3.3×10^6 psi). This material was used as the spring material because it can store more energy per pound of spring than any metal. Results of the study are shown in Figures 78 through 81. These curves show the characteristics of a cantilevered hollow tube, circular in cross section, and tapered in diameter such that the tip diameter is half the root diameter. Tube wall thickness is 0.100 inch which was selected as the minimum thickness for this material to achieve the high allowable stresses. The two-to-one taper in diameter was chosen for minimum weight while still maintaining some bending strength at the tip. It should be noted that the calculations apply to a horizontal cantilever strut with a vertical ground reaction at the tip. In actual practice the strut would be curved down at the tip, to achieve a desired ground clearance for the vehicle, and this would result in curves of the same trends but slightly different numerical values. One calculation of a curved strut was made to show that the spring rate could be achieved for a curved strut with the same horizontal projected cantilever length and the same root diameter as a straight strut. The weight of the curved strut is slightly heavier than the straight strut due to its greater developed length.

Strut deflection in terms of landing velocity and maximum landing load factor is shown in Figure 78. This curve is for a spring with constant spring rate, and is independent of all other spring characteristics, so that it applies equally well to a straight or a curved strut. It can be seen that for low load factors and high velocities the deflections become large compared to the size of the vehicles. Conversely for small deflections at high velocities it is necessary to allow high load factors. Practical maximum values have been imposed on each parameter as depicted by the dashed shaded line: 12 inches maximum deflection which is reasonable for vehicles of this size, and 8 earth g maximum load factor which is identical to the maximum load factor imposed on the vehicle by the LEM truck. Although 8 is a high load factor for a manned vehicle it is well within the tolerance level of the human anatomy for short time impacts. Landing gear design should be restricted to areas to the left of the 8 g limit line and below the 12 inches deflection line.

Cantilever strut weight is shown on Figure 79 for various vehicle landing weights. It is interesting to note that the weight of strut for any specific vehicle weight is a function of the landing velocity only. This follows from the capability of the material to absorb a specific amount of foot-pounds of energy per pound of material. The dashed limit line on Figure 79 indicates the maximum capability possible within the limits established on load factor and deflection shown on Figure 78.

Cantilever strut length and diameter are plotted on Figures 80 and 81 as a function of vehicle weight, load factor, and landing velocity. The limits of Figure 78 are also superimposed on these figures for a vehicle weight of 1400 pounds. After comparing the weight, size, and proportions of struts designed for various landing velocities it was decided that all vehicles would incorporate spring struts capable of landing with 10 feet per second at 8 g. As previously discussed this implied a performance

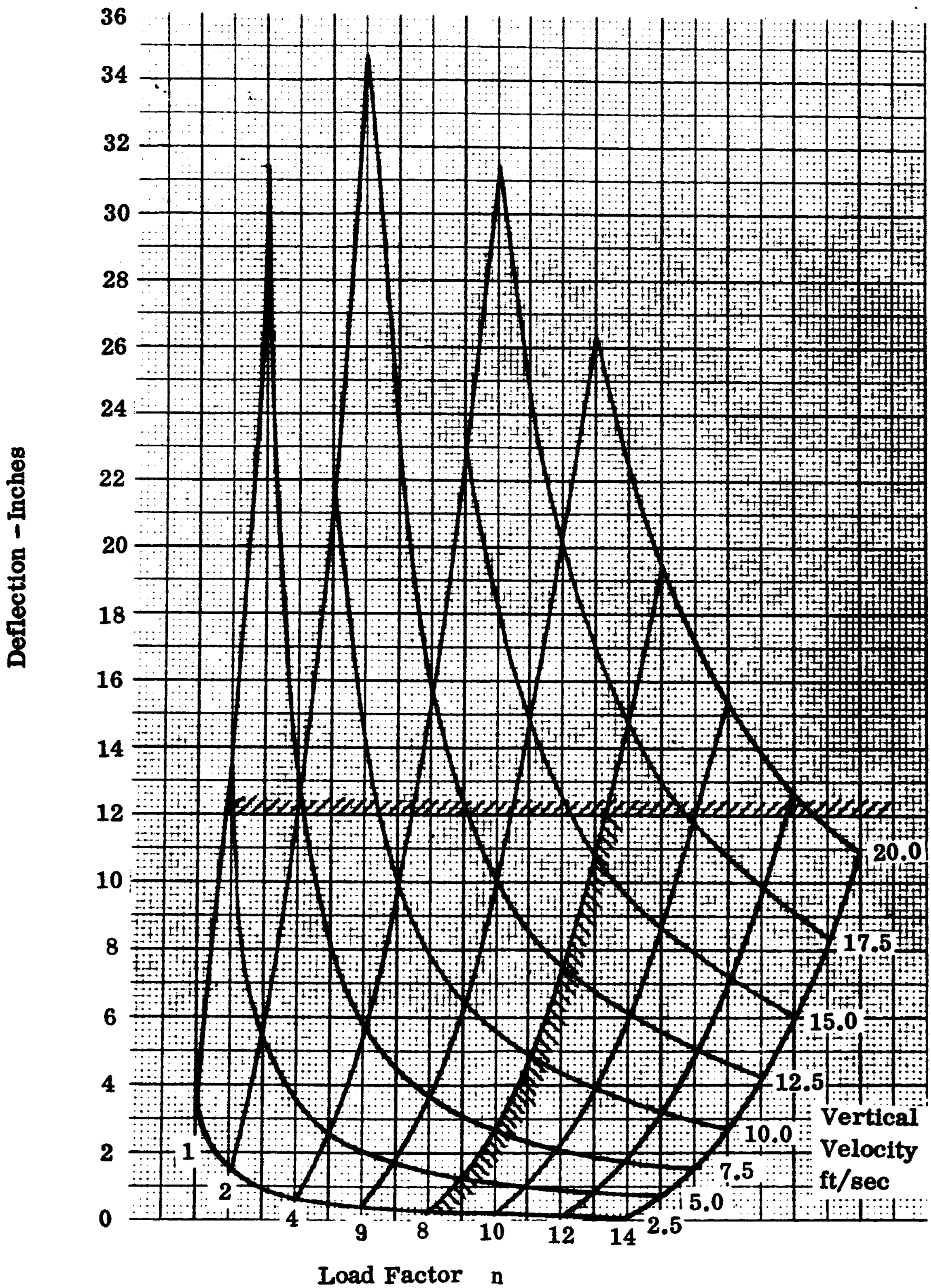


Figure 78. Landing Gear Deflection

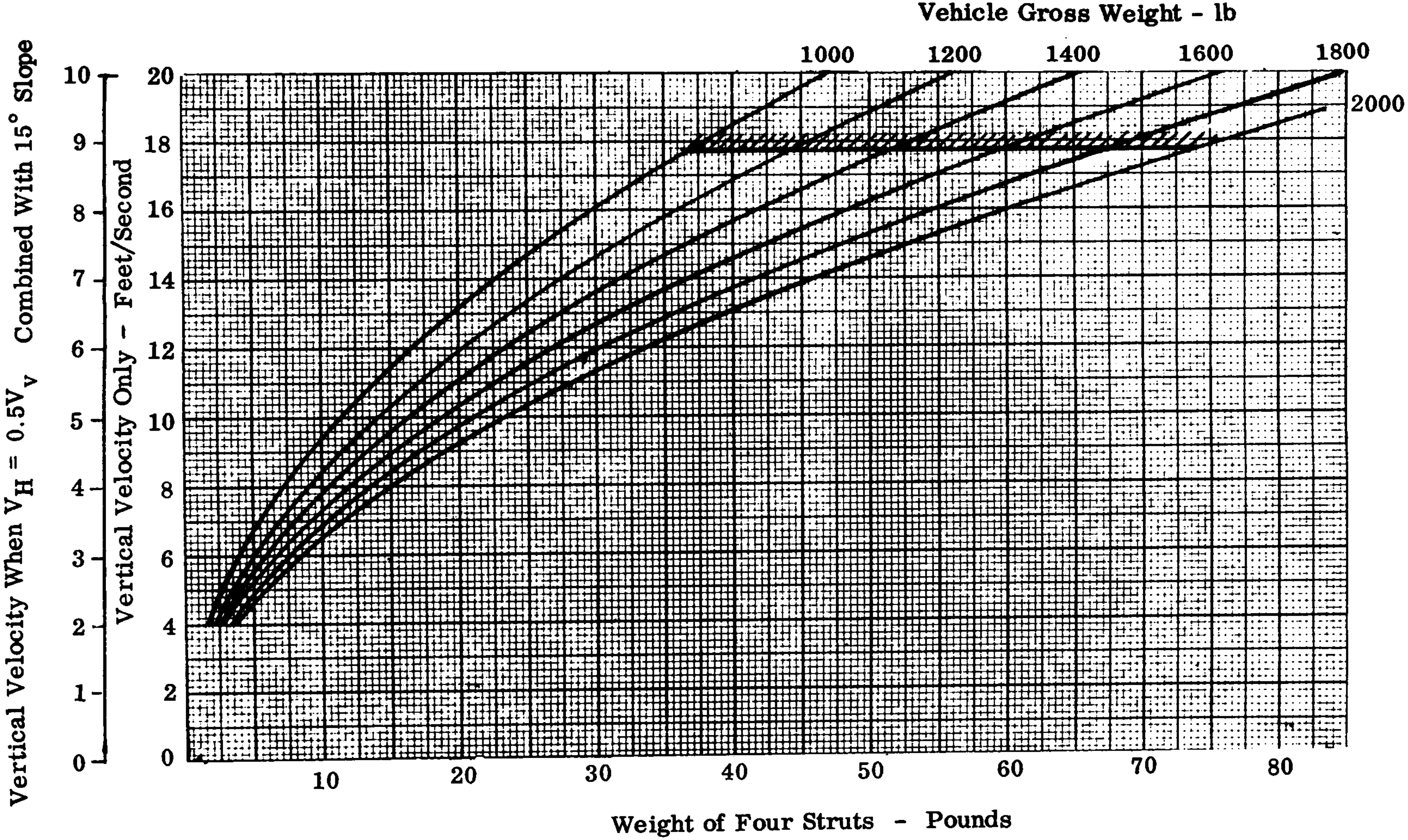


Figure 79. Fiberglass Spring Gear Weight for Several Vehicle Weights

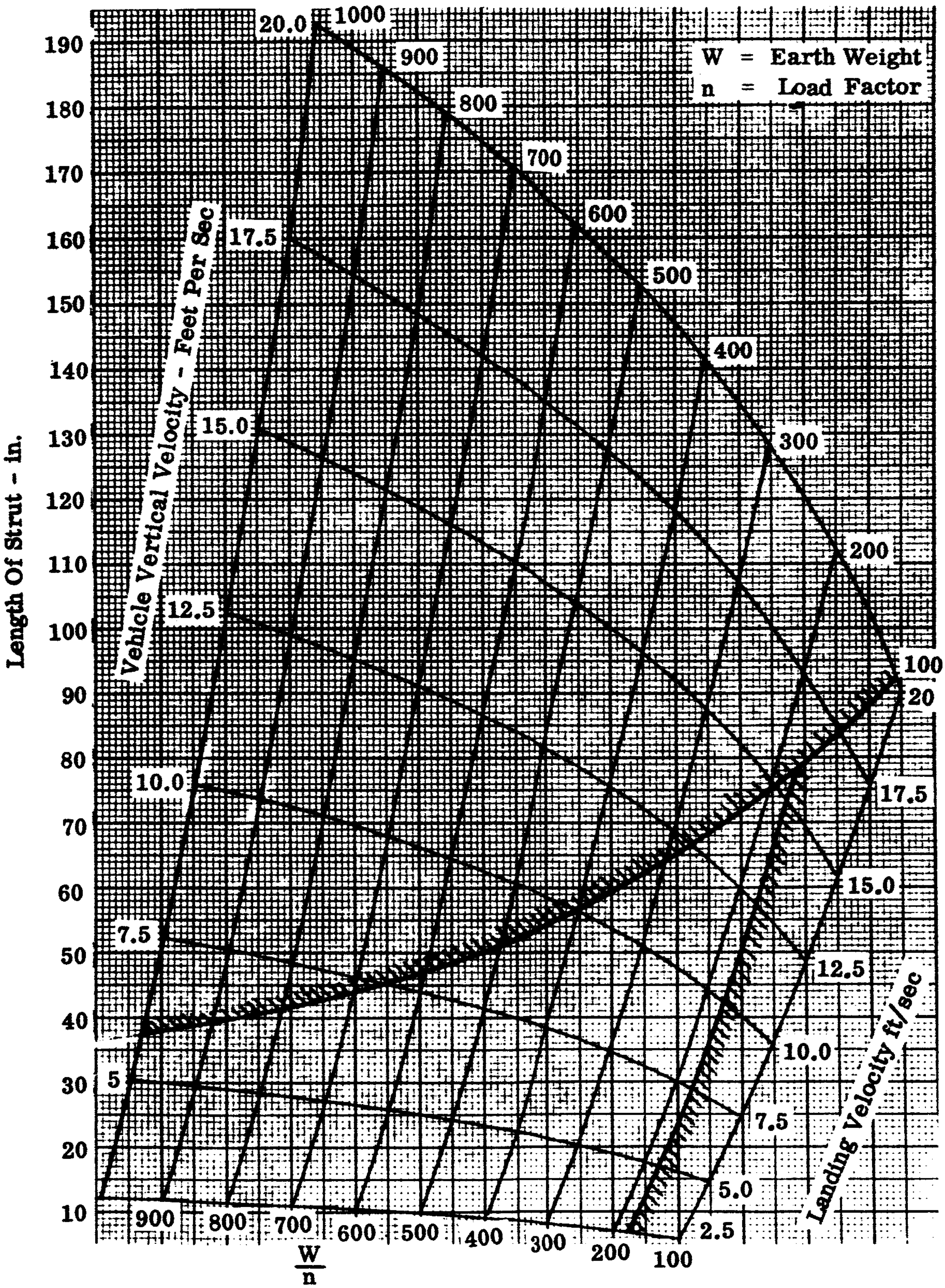


Figure 80. Length of Strut for Vertical Landing on Four Horizontal Bending Spring Landing Gear Struts

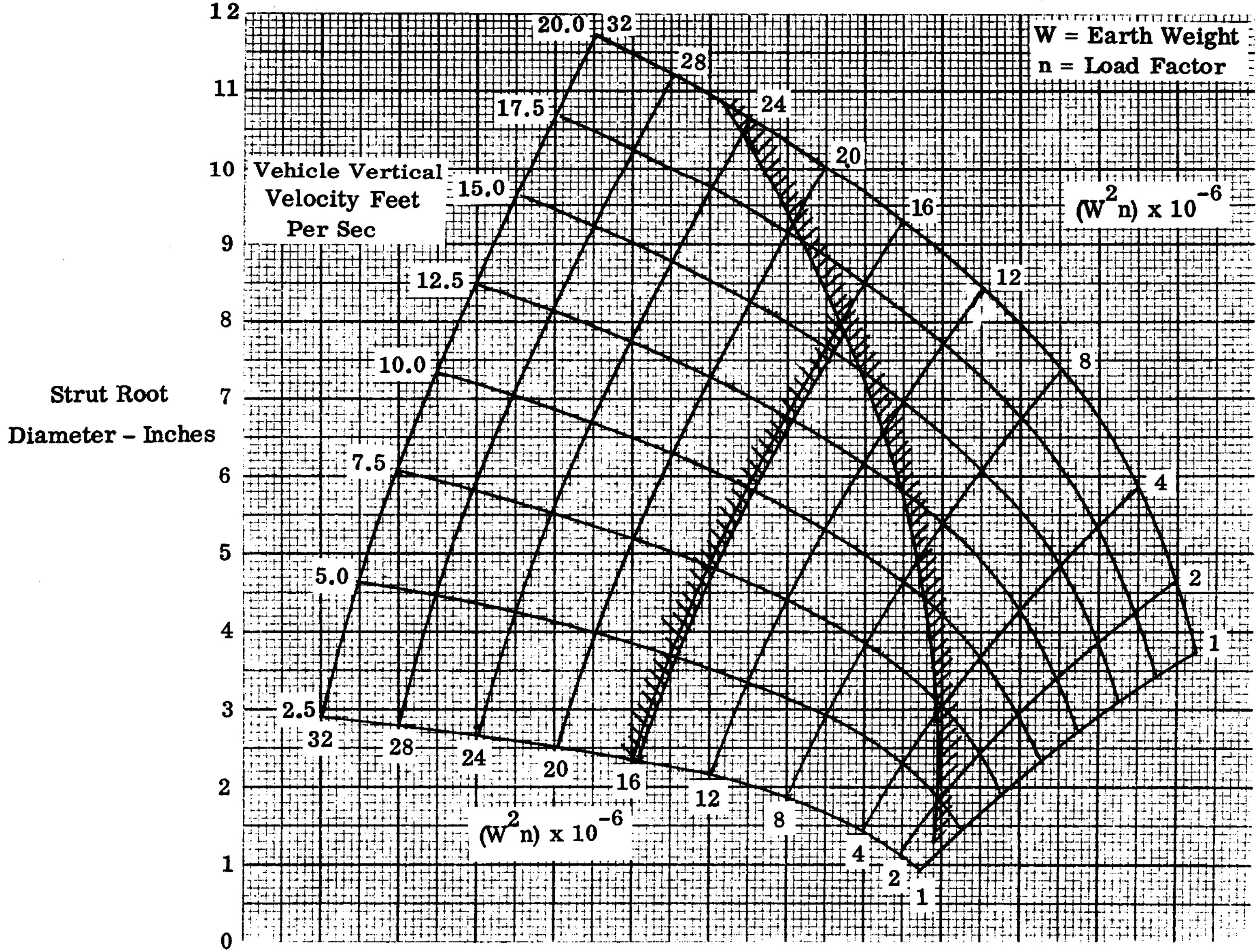


Figure 81. Root Diameter of Strut for Vertical Landing on Four Horizontal Bending Spring Landing Gear Struts with Tip Dia. = 1/2 Root Dia

envelope of 5 feet per second vertical velocity and 2.5 feet per second horizontal velocity.

A spring type landing gear has two distinct advantages over other types of landing gears: it is completely reusable, and it has no moving parts. It also has two distinct disadvantages: it has no internal damping, and it becomes too large and unwieldy and unstable when designed for large landing velocities. The advantages of the spring gear are attractive enough that it merits further study to see if the disadvantages could be overcome. One area of study is the landing velocity criteria. Helicopter experience shows that given sufficient control a man can land a vehicle at nearly zero velocity. Methods of control and guidance should be sought to give the pilot the same precision of maneuver in a lunar vehicle that is possible with a helicopter. Another area of study is the external damping of a spring gear due to scrubbing of the landing foot on the lunar surface when deflecting. This is an area that depends on the characteristics of the lunar surface and nothing definite can be concluded in this regard until the surface characteristics are defined, but predictions can be made based on a variety of assumed surfaces. It may also be possible to develop a technique in the guidance and control area which would overcome the rebound characteristic of a spring gear. For instance, "power on" landings would permit the pilot to make another approach after rebound if he failed to achieve a near zero velocity surface contact on the first approach.

Parametric investigation of shock absorbing gears has already been performed in previous studies as previously pointed out in the text. In the simple point design which was investigated for comparison with the spring type gears, the gear weight was 30 pounds heavier than the spring gear for a 1400 pound gross weight vehicle. The second disadvantage of the crushable gear is the need for replacement of the energy absorption material after a hard landing. This represents a serious disadvantage especially when it has to be done by a single astronaut away from the lunar base.

C. THERMAL CONTROL

Vehicle performance and reliability are directly related to the thermal history during the mission, it is therefore of primary importance to predict the temperature excursions experienced by the various subsystem components; e.g., propulsion system propellants, pressurization gas and the battery electrolyte.

The thermal control of a vehicle and/or systems on the surface of the moon is complicated by the extremes in temperature and incident radiation encountered. It has been shown in Reference 2 that the lunar surface temperature reaches 240°F or more at the subsolar point and during lunar night drops to -240°F or lower as a result of the absence of an atmosphere to attenuate or diffuse direct solar radiation.

For purposes of comparison, a basic configuration, i.e., a two man escape vehicle was selected for a preliminary investigation of the influence of lunar environment on the thermal design. It should be noted that the results are applicable to any vehicle within the limits of the assumptions listed below:

1. Lunar Environment

   The temperature of the lunar surface at the subsolar point is 240°F.

   The temperature of the lunar surface on the dark side is -240°F.

   The thermal conductivity of the lunar surface is negligible.

   The spatial environment is considered to be a radiation sink at -460°F (0° absolute).

   The lunar surface emits and reflects radiation diffusely in accordance with Lambert's law.

   The reflectance of the lunar surface is the same over the entire surface.

   The lunar surface in the immediate vicinity of the vehicle may be analytically approximated by an infinite flat plane of uniform temperature.

   During the lunar daylight period, the absolute temperature of the lunar surface is expressed as a function of the time.

   The ecliptic plane and the lunar equatorial plane are coplanar.

2. Vehicle and/or Propellant Tank

   The surface of the vehicle reflects and emits radiation diffusely in accordance with Lambert's law.

   The absorptance of the surface of the vehicle to infrared radiation is equal to its emittance (i.e., exhibits gray-body characteristics in the infrared region).

   Radiation that is reflected or emitted from the vehicle does not return to its surface (except as indicated for the radiation-shield analysis).

   The outer skin of the vehicle and/or the propellant storage tank has an infinite thermal conductivity in a direction orthogonal to the flow of heat. Hence, the surface temperature of the storage tank is uniform at any given time. It is noted that considerable variations of temperature may exist over the vehicle surface; however, it was felt that an average temperature would provide a representative temperature as a function of time. It is shown in Reference 4 that the error introduced by this assumption produces slightly conservative results with respect to the amount of heat entering or leaving the propellant, i.e., less heat actually enteres or leaves the propellant.

The temperature of the surface of the vehicle is assumed to be that reached at equilibrium.

No heat is lost from the surface of a vehicle by conduction or radiation into or through the vehicle.

No heat is conducted into the vehicle from the lunar surface or vice versa. (The vehicle is on a nonconducting base.)

The shadow cast upon the lunar surface by the vehicle has a negligible effect upon vehicle temperatures.

Based on these assumptions, a heat balance was written and solved. The heat balance includes inputs of direct solar radiation, solar radiation reflected from the lunar surface and radiation emitted from the lunar surface which are equated to the heat radiated from the system and the internal energy of the system, i.e., the thermal capacity.

A transient analysis was conducted to determine the response of the propellant bulk temperature. It was assumed that oxidizer (N_2O_4) was contained in a thin-walled tank with a surface coating representative of gold plate on aluminum, i.e., $\alpha_s/\epsilon = 10$, $\epsilon = 0.03$. Since the oxidizer has a lower thermal capacity, i.e., product of mass and specific weight than the fuel (50% hydrazine - 50% UDMH) for an equivalent volume, the analysis is somewhat conservative with respect to the fuel. The results are shown in Figure 82 for one lunar day. It is obvious that further damping of the periodic heating is required since the mean bulk temperature and the double amplitude of the temperature excursion greatly exceeds the allowable temperature range of the propellants: 30°F to 100°F.

As a means of evaluating the practicality of insulation employed to damp the periodic temperature change of the oxidizer to an acceptable level, it may be assumed that the insulated tank wall is analogous to a semi-infinite thick flat plate whose outer surface temperature varies periodically. A classical solution from Reference 3 was employed to determine the amplitude of the temperature wave for various depths of penetration. The results are shown in Table VIII. It was found that at an insulation depth of 1 inch (Linde Co. SI-91) the temperature wave is damped to approximately 11% of the outer surface value. In addition, the net heat flow rate through the insulation during the positive half of the cycle was integrated to determine the temperature response of the oxidizer. The results, also shown in Table VIII, are presented in normalized form as N_2O_4 bulk temperature rise times weight per unit area. Assuming 200 lb of N_2O_4*, the oxidizer bulk temperature will cycle approximately ±8°F about an undefined mean temperature if the tank is insulated with 0.25 in. of Linde Co. SI-91. Therefore, it can be concluded that insulating the propellant tank is feasible.

* Selected as a typical loading in one of two oxidizer tanks

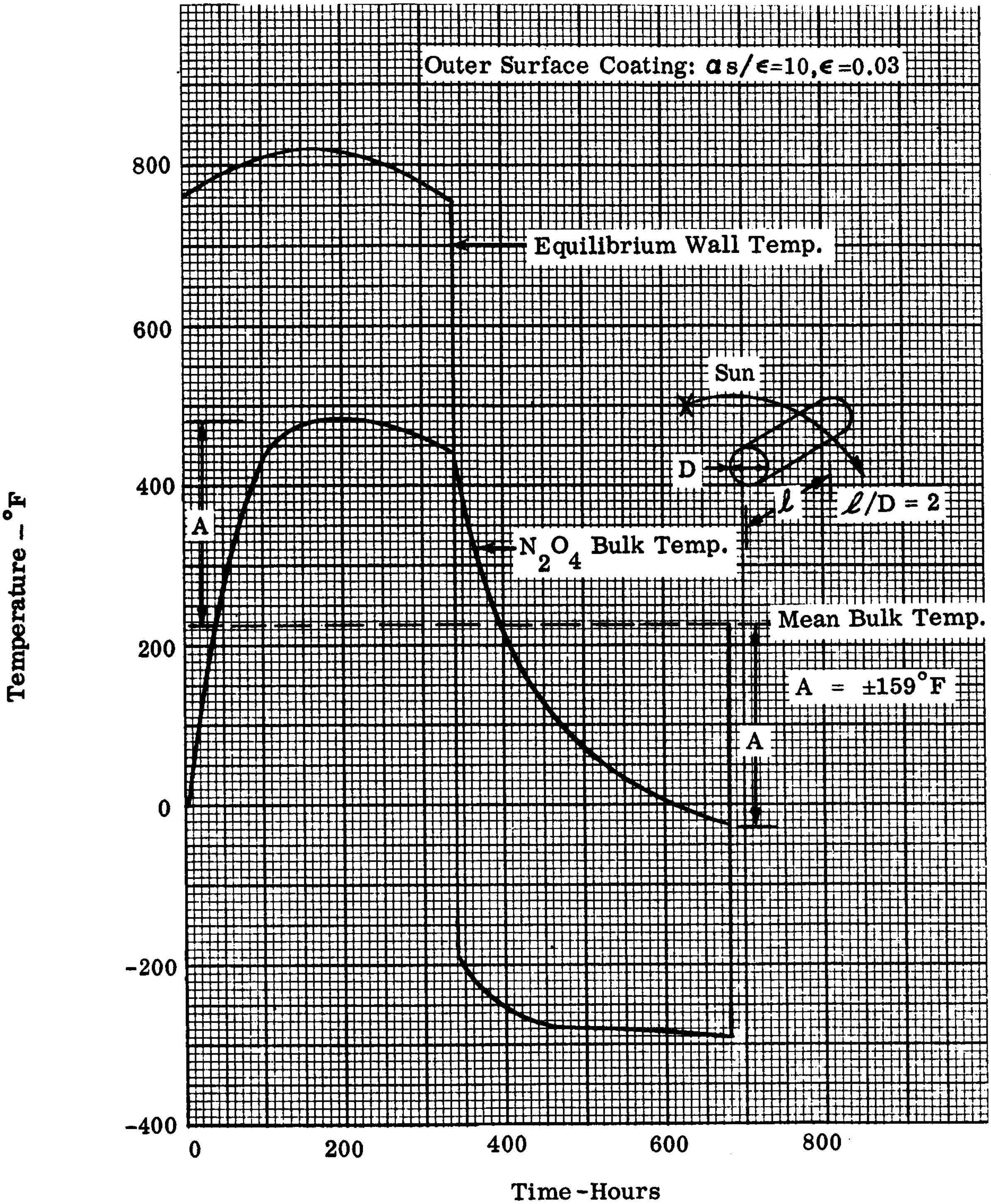


Figure 82. Transient N_2O_4 Bulk Temperature During One Lunar Day (28.5 Days) No Insulation

As a further refinement, computer techniques were employed to determine the oxidizer temperature response for various insulated tank designs. The oxidizer transient bulk temperature during one lunar day was determined for a cylindrical tank (length/diameter ratio = 2) insulated with various thicknesses of Linde Co. SI-91 superinsulation and outer surface coatings. The results presented in Figures 83 and 84 indicate that a thickness of 0.25 in. of SI-91 insulation is sufficient to constrain the oxidizer bulk temperature within the design range, i.e., ±35°F, and that an outer surface coating with $1 < \alpha_s/\epsilon < 10$ is required to achieve the desired mean bulk temperature. It should be noted that the analysis presumes that the propellant tank is unshielded by the vehicle and is oriented to receive maximum solar heating, which is quite reasonable for the truss type structure selected for these vehicles.

In conclusion, a thermal design employing a combination of multilayer insulation and surface coating is feasible to maintain stabilized temperatures within the prescribed limits during a 180 day storage on the lunar surface

TABLE VIII

AMPLITUDE OF THE OXIDIZER BULK TEMPERATURE EXCURSION IN RESPONSE TO A SINUSOIDAL VARIATION OF INSULATION OUTER SURFACE TEMPERATURE (±1000°F)

Thermal capacity of N_2O_4 = 0.37 BTU/lb°F
Thermal diffusity of SI-91 = 6.35×10^{-6} ft^2/hr

Depth of Penetration (Thickness) ~ in.	Amplitude of Temperature Wave ~ °F	N_2O_4 ΔT_b (W/A_s) ~ °F(lb/ft^2)
0.25	568	125.9
0.50	324	71.6
1.00	104	23.0
2.00	10.5	2.3

D. MICROMETEORITE PROTECTION

The Micrometeorite protection requirements for the Personnel Propulsion Devices are presented in two phases: (1) protection of the sensitive components of the vehicle during storage period on the moon and (2) protection of the man during operation of the vehicle. Because the exposure time in each of these phases vary significantly, the requirements for protection are treated separately.

The most straight forward method of protecting a vehicle or its components from meteoroid penetration would be to provide an outer skin of sufficient thickness to about the impact energy of meteoroids encountered during its operational and storage periods in the hostile environment. A more sophisticated approach, which contributes less weight to the overall system is to provide a number of "bumpers"

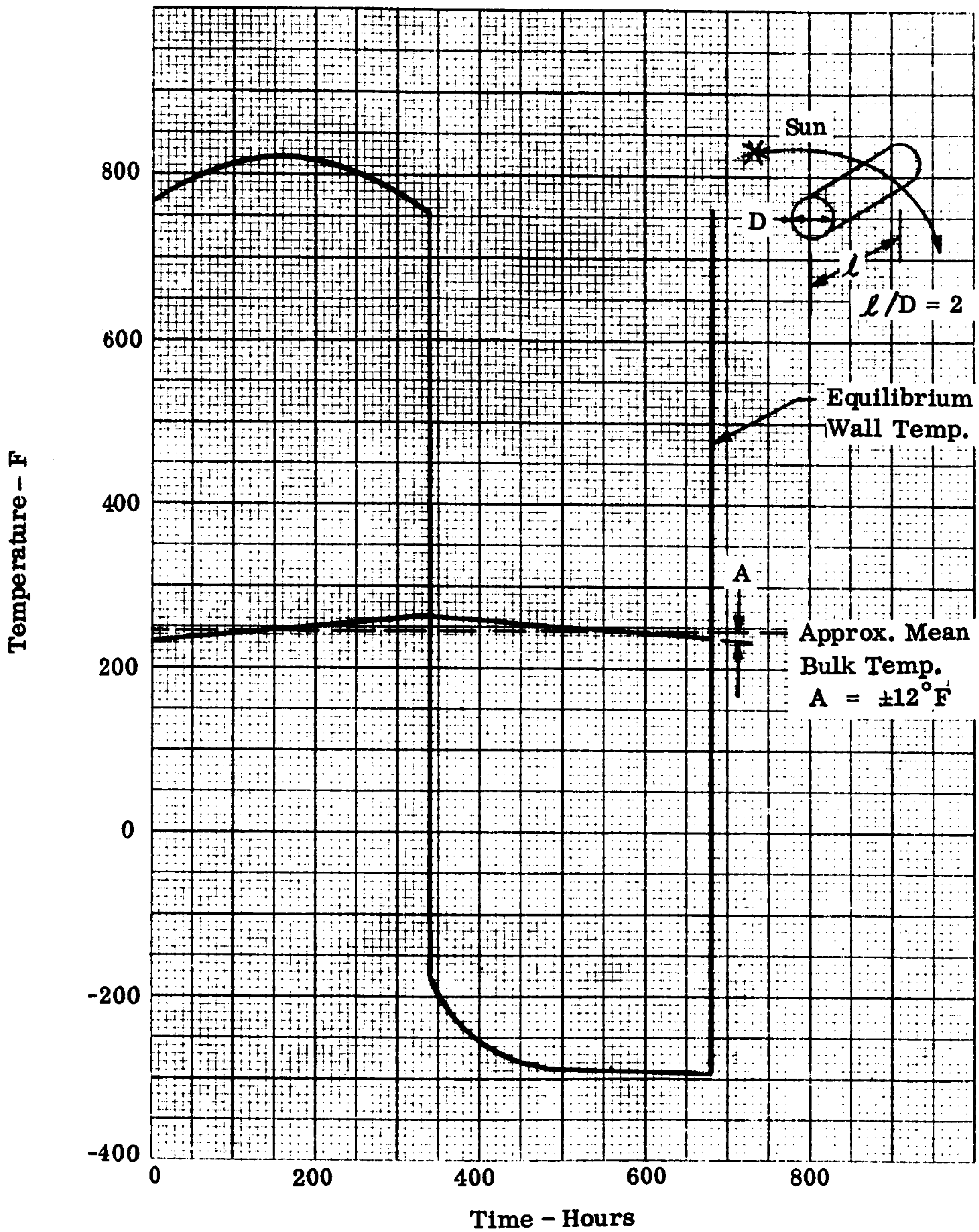


Figure 83. Transient N_2O_4 Bulk Temperature During One Lunar Day (28.5 Days) Insulated with 1/4 Inch of SI-91 Outer Surface Coating: $\alpha_s/\epsilon = 10.0$

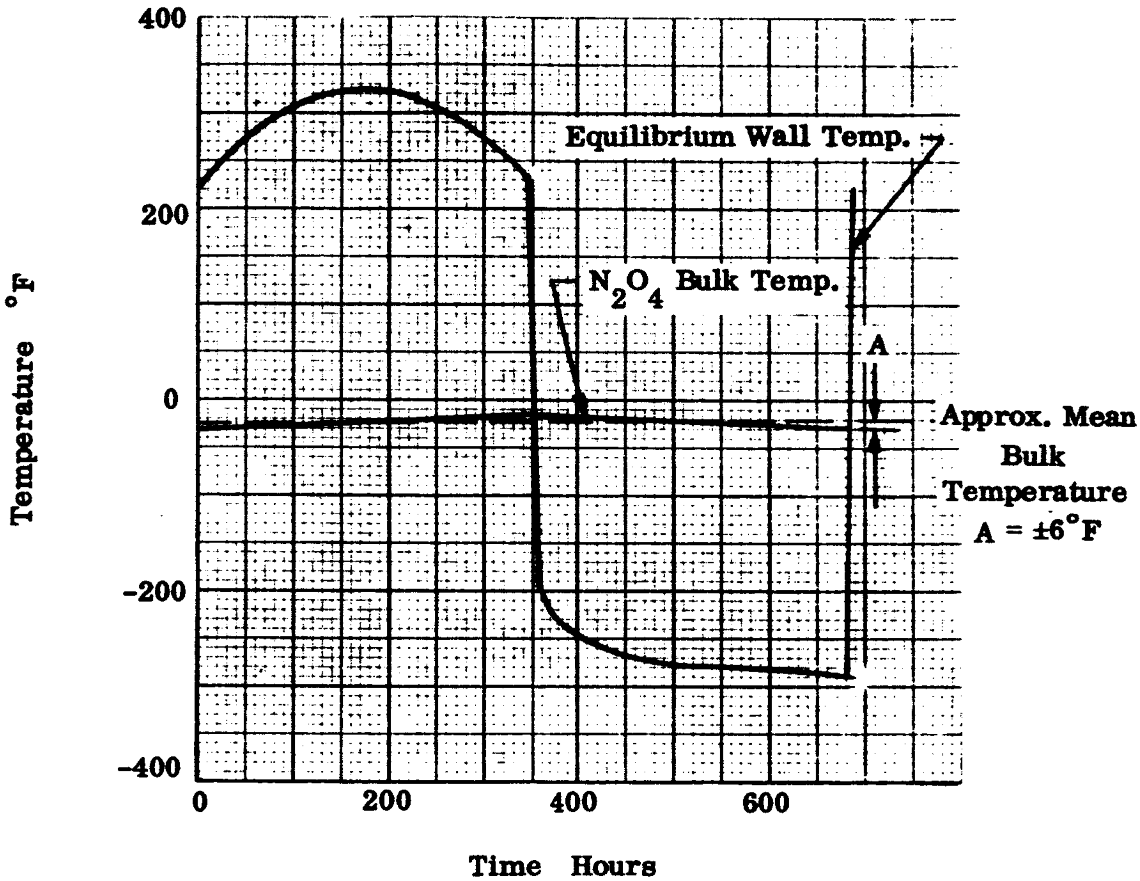


Figure 84. Transient N_2O_4 Bulk Temperature During One Lunar Day (28.5 Days) Insulated with 1/4 Inch of SI-91 Outer Surface Coating: α_s/ϵ = 1.0

spaced a distance apart. A third method is to functionally integrate the meteorite shield with the insulation which is externally applied to the propellant and pressurant tanks for thermal control. These three methods were evaluated for a 180 day storage period using meteoroid flux and puncture models presented in Reference 5, which are considered to be the most realistic at the present time.

Reference 5 presents a cumulative flux model of primary meteoroids at low to median latitude on the lunar surface for meteoroid masses between 10^{-10} and 1 gram and a puncture model for vehicles or structures with homogeneous metallic walls. The number $F_{>m}$ of meteoroids per square meter second with mass equal to or greater than m grams is related to m by:

$$\log F_{>m} = (-\log m) - 14.58 \pm 1.35$$

The puncture model from primary meteoroids with random orientation on the lunar surface is defined as

$$\log P = 1/3 \left[\log (At) - \log \log (1/R) - \log (\rho_t H_t) \right]$$

where
- P = wall thickness punctured by meteoroids of mass m, (cm)
- A = effectively exposed area, (m^2)
- R = probability that area A will not be punctured in time (t)
- H_t = wall hardness in Brinell units
- t = exposure time, seconds
- ρ_t = wall density, g/cc

For a single wall aluminum (2219-T87) structure and a no-puncture probability of 0.99 with a confidence level of 80%, the primary meteoroid puncture model becomes

$$\log P = 1/3 \log (At) - 3.035$$

where
- H_t = 128
- ρ_t = 2.82 g/cc

The above equation is graphically shown in Figure 85. Table IX summarizes the minimum structure requirements for the vehicle stored for 180 days on the lunar surface. The exposed areas in Table IX refer to the critical areas of the vehicle which will be exposed to micrometeorite impacts such as pressurant and propellant tanks, instrumentation and plumbing. Considering 0.5 inch of super-insulation (Linde SI-91) around the propellant tanks, the total equivalent aluminum thickness is 0.100 inch: 50 layers of 0.00023 aluminum foil and 50 layers of 0.0025 glass paper. The exposed surfaces of the tanks amount to approximately 20 ft^2. Examination of Table IX indicates that 0.100 inch aluminum is more than sufficient to protect the

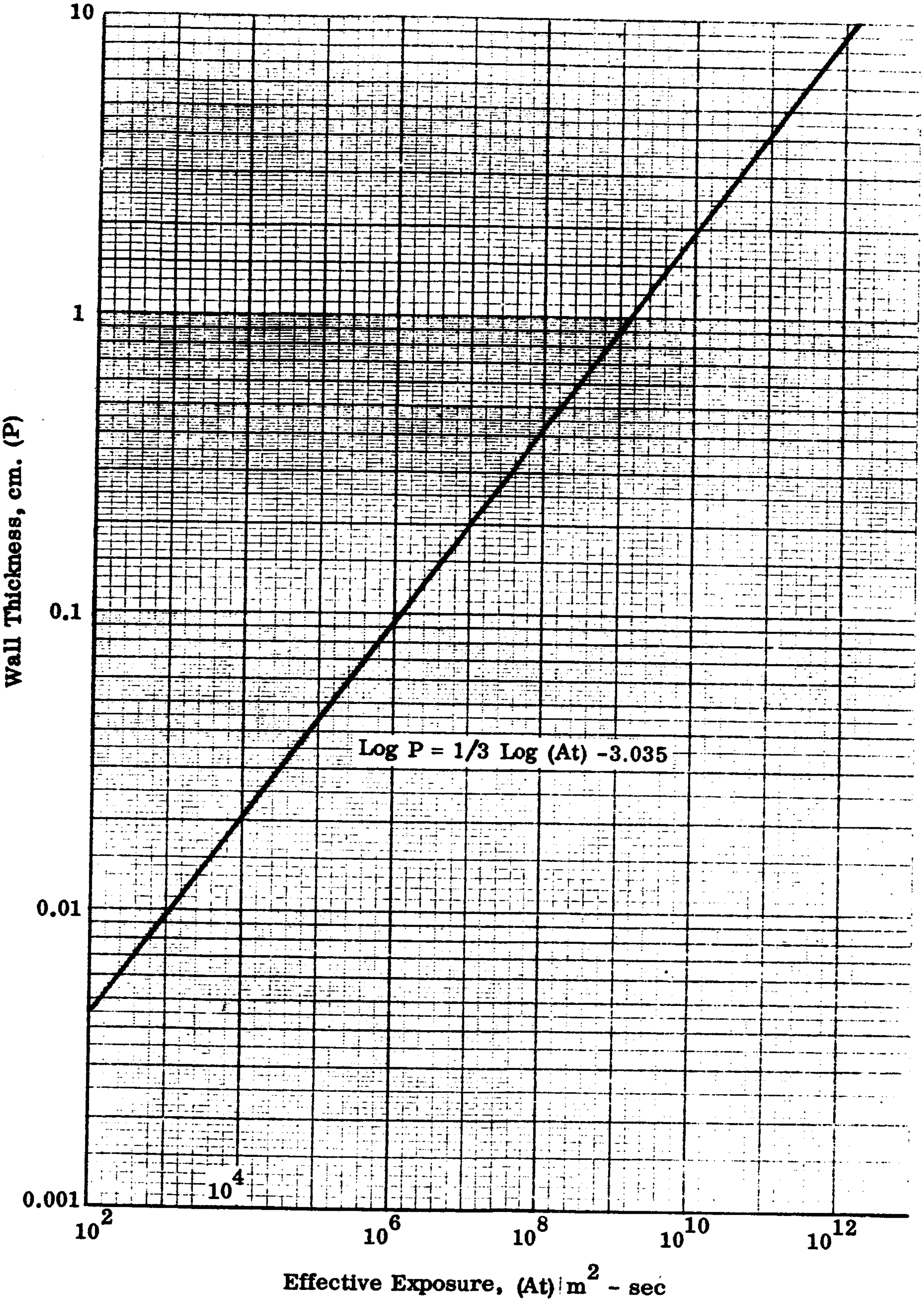


Figure 85. Aluminum Wall Thickness Required for a No-Puncture Probability 0.99 and a Confidence Level of 80%

tanks from micrometeorite impact with a no-puncture probability of 0.99 and a confidence level of 80%. Since the multilayer bumper effect of the insulation was not taken into consideration, this is a very conservative conclusion.

TABLE IX

THICKNESS OF ALUMINUM REQUIRED ON VEHICLE FOR A NO-PUNCTURE PROBABILITY OF 0.99 AND CONFIDENCE LEVEL OF 80%

		Aluminum Thickness		
			Double-Wall	
Exposed Area (ft^2)	Exposed Time (Days)	Single-Wall (in.)	Under-Wall (in.)	Bumper (in.)
1	180	0.036	0.0092	0.0031
5	↑	0.066	0.017	0.0056
10		0.076	0.020	0.0066
15		0.084	0.021	0.0072
20		0.090	0.023	0.0076
25		0.096	0.025	0.0082
30		0.105	0.027	0.0090
35	↓	0.112	0.029	0.0096
40	180	0.122	0.032	0.0100

Meteoroid Flux: Log F = - log m - 14.58 ±1.35 (MSFC 4-22-64)
F = impacts/m^2-sec with mass equal to or greater than m grams

Extravehicular activity on the lunar surface requires an external thermal garment over the space suit consisting of a material such as multilayer aluminized mylar (NRC-2) insulation. A single layer of aluminized Mylar is 0.00025 inch thick and has an equivalent aluminum thickness of about 0.000134 inch (see Table X). Considering 30 ft^2 of exposure area on the space suit, Table XI indicates the allowable exposure times for insulation thickness of 0.10, 0.25 and 0.50 inch. Ten layers of NRC-2 (0.10 inch) will permit 26 minutes of exposure with a no-puncture probability of 0.999 and 80% confidence level. Since the typical mission duration of a surface translation mission is approximately 20 minutes, and the insulation thickness will probably be a minimum of 0.25 inches, no other micrometeorite shield will be required in addition to the thermal insulation during flight. As shown in Table X, 0.25 inch of insulation will permit 6.75 hours of exposure. A conservatism in this analysis is also introduced by neglecting the additional protection provided by the multilayer bumper effect of the insulation.

In summary, the insulation over the propellant tanks and the space suit provide adequate shielding against micrometeorite impacts. Although insulation may not be required over exposed instrumentation and plumbing, the use of insulation will protect these components against micrometeorites in a similar manner.

TABLE X

ALUMINUM EQUIVALENT THICKNESS OF NRC-2 INSULATION

No. of Layers	Total Mylar Thickness (in.)	Aluminum Equivalent Thickness (in.)
1	0.00025	0.000134
10	0.025	0.0134
20	0.050	0.0268
30	0.075	0.0402
40	0.100	0.0536
50	0.125	0.0670
60	0.150	0.0804
70	0.175	0.0938
80	0.200	0.1072
90	0.225	0.1206
100	0.250	0.1340

TABLE XI

NRC-2 INSULATION FOR METEOROID SHIELDING OVER SPACESUIT

Thickness (in.)	Layers	Alum. Equivalent Thickness (in.)	Allowable Exposure Time
0.10	10	0.0134	26 min
0.25	25	0.0335	6.75 hr
0.50	50	0.0670	54 hr

0.999 No-Puncture Probability

80% Confidence level

VII. CONFIGURATION STUDIES

A. SUMMARY

The configurations presented in this section constitute those which have survived the screening process completed during the course of the study. Many other configurations were investigated but are not shown because of their elimination early in the study. The data presented for the selected configurations is oriented so as to provide inputs for the simulation studies. Summary of weight, center of gravity excursions and moments of inertia have been tabulated for all configurations which have been submitted for simulation. In those cases where center of gravity excursions were large, and moments of inertia followed a nonlinear trend with propellant consumption (i.e., in the back pack configurations), a graphical presentation of the variations is included.

In general, the tables listing the pertinent characteristics for each group of vehicles should be adequate to initiate simulation studies. If during these studies modifications to the configurations submitted become necessary to improve their operational and/or handling characteristics, then the parametric data presented in the previous sections may be used to provide some of the necessary information.

While it was outside the scope of this contract to undertake design studies in the subsystem areas, a brief survey of available equipment which meet the vehicle requirements was completed in order to define reasonable weights upon which the vehicle nominal propulsive capability could be defined and vehicle design characteristics established. The subsystems so defined for the Personnel Propulsion Devices are as follows:

(1) An automatic checkout system to display the systems' operational readiness.

(2) A center of gravity sensor to permit alignment of the vehicle c.g. with the thrust vector within tolerable limits.

(3) A communications transceiver to permit line-of-sight communications between the Personnel Propulsion Devices and the launch site on the ground, or the lunar base or an orbiting spacecraft.

(4) A simple sighting device or low power telescope to provide visual alignment between the vehicle and the target.

(5) The necessary drive, logic and control circuitry required to drive the throttle valves used in the propulsion system.

The nominal weights and power consumption of the equipment required to perform the above functions have been presented in Section V in groups of varying complexity and sophistication. The equipment weight and power supply incorporated in each vehicle configuration presented in this section corresponds to the simplest group of equipment presented in Section V. Modifications to the vehicles by substitution of equipment to augment their characteristics can be made as indicated by the results of the simulation studies.

B. CONFIGURATION GROUND RULES AND SCREENING PROCESS

The primary mission objectives of the Personnel Propulsion Devices are two-fold: (1) to provide transportation for lunar-based personnel between two or more points on the moon and (2) to provide escape capability from the surface of the moon into lunar orbit followed by orbital rendezvous and subsequent docking with the orbiting spacecraft. The predominating design consideration on these vehicles, as utilized in this study, is simplicity with emphasis placed on maximum utilization of the astronaut's capability. The following general ground rules were observed during the generation of the conceptual vehicle configurations presented in this section of the report:

(1) Extravehicular space suits designed for the lunar missions will be available.

(2) A cabin enclosure for the flight is not required.

(3) The propellants established for LEM (N_2O_4/50 UDMH-50 N_2H_4) would receive primary consideration for the main propulsion system.

(4) Lunar refueling operations are permissible and propellants and pressurant supplies are available from lunar stores.

(5) Transportation vehicles are reusable with a minimum of maintenance; escape vehicles have a single mission capability.

(6) The equipment initially installed on the vehicle would be limited to the minimum required for control, guidance and navigation.

(7) Present state-of-the-art technology would be employed with extrapolations limited to the 1968-1970 time period.

(8) Personnel Propulsion Devices would be stored in the vehicle which transported them to the moon until the time of their intended use.

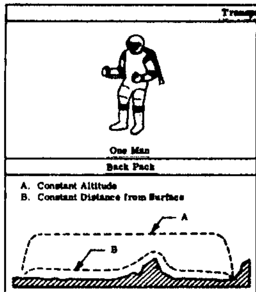
(9) Two-man vehicles would be piloted by only one man.

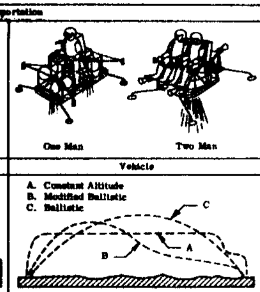
Based upon these ground rules a series of one- and two-man conceptual configurations have been generated which permit NASA, through simulation studies, to conduct performance/controllability/weight evaluation of minimum complexity devices for use in the vicinity of the moon.

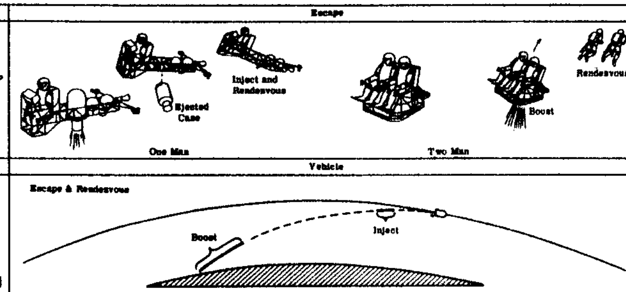
The requirements for sizing the basic configurations are shown in Table XII. Included in this table are transportation devices of varying range capability provided in terms of nominal Δ V increments, escape devices of varying thrust to initial weight ratios, and dual function devices of fixed thrust level and Δ V capability. Additional variations have been provided within selected configurations to enable evaluation of specific design features. For instance, single and multiple engine

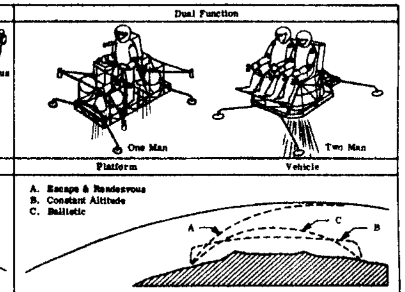
TABLE XII

APPLICABLE CONFIGURATIONS









- Short Range Transportation over Rough or Smooth Terrain - Constant Thrust Application Opposing Gravity
- Investigation of Otherwise Inaccessible Areas (Craters, etc.)
- Multiple Mission Capability

- Intermediate Range Transportation
- Constant Thrust Application Opposing Gravity
- Ballistic or Modified Ballistic

- Escape into 80-100 n.mi. Orbit
- Single Mission Capability

- Escape & Rendezvous
- Long Range Transportation
- Less Than 15° Slope Required on Landing Area
- Multiple Mission Capability

	Back Pack	Back Pack	1 M. Vehicle	2 Man Vehicle	2 Man Vehicle	2 Man Vehicle	1 Man Vehicle	1 Man Vehicle	1 Man Vehicle	2 Man Vehicle	2 Man Vehicle	2 Man Vehicle	1 Man Vehicle	1 Man Vehicle	2 Man Vehicle	One Man Platform	2 Man Vehicle
Transporation	x	x	x	x	x	x										x	x
Escape							x	x	x	x	x	x	x	x	x	x	x
Thrust Level (T/W_E)	0.5	0.5	0.5	0.5	0.5	0.5	0.2	0.5	1.0	0.2	0.5	1.0	0.5	1.0	1.0	0.5	0.5
Throttling Range	10:1	10:1	10:1	10:1	10:1	10:1	-	-	-	-	-	-	-	-	-	10:1	10:1
Nonpropulsive Wt (est) Man	245	245	245	490	490	490	245	245	245	490	490	490				245	490
Payload	30	30	30	-	-	-	30	30	30	-	-	-	30	30	-	30	
Electronics	42	42	42	42	42	42	42	42	42	42	42	42	42	42		42	42
Attitude Control (approx.)	-		40	40	40	40	40	40	40	40	40	40	40	40		40	40
No. of Chamber Tanks	- -		6/12 Main	6/12 Main	6/12 Main	6/12 Main	6/12 Main	6/12 Main	6/12 Main	6/12 Main	6/12 Main	6/12 Main	6/12 Main	6/12 Main	6/12 Auxiliary	6/12 Main	6/12 Main
Structure	20	30	70	60	70	80	75	75	75	80	80	80	80	80		75	80
Landing Gear	-	-	30	35	45	50	-	-	-	-	-	-	-	-	-	30	50
ΔV (fps) (Ideal)	2000	4000	6000	2000	4000	6000	8000	8000	8000	8000	8000	8000	8000	8000	8000	8000	8000
No. of Tanks (1)	4 sph	4 sph	4 sph	4 sph	4 sph	4 sph.	4 sph	4 sph	4 sph	4 sph	4 sph	4 sph	2 sph	2 sph	BACK PACK SECOND STAGE	4 sph	4 sph
(2)	3 cyl	3 cyl	4 cyl	4 cyl	4 cyl	4 cyl	4 cyl	4 cyl	4 cyl	4 cyl	4 cyl	4 cyl	-	-		4 cyl	4 cyl
(3)	2 sph	2 sph	2 sph	2 sph	2 sph	2 sph	2 sph	2 sph	2 sph	2 sph	2 sph	2 shp	-	-		2 sph	2 sph
(4)	2 cyl	2 cyl	2 cyl	2 cyl	2 cyl	2 cyl	2 cyl	2 cyl	2 cyl	2 cyl	2 cyl	2 cyl	-	-		2 cyl	2 cyl
No. of Chambers (F=Fixed) (G=Gimballed)	2G	2G	1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F	1F - - - - -	1F - - - - -		1F 1G 2F 2G 4G 4F	1F 1G 2F 2G 4G 4F
Pressurization System Duty Cycle	Reg He	Reg He	Reg He	Reg He	Reg He	Reg He	Reg He 100% F_{max}	Reg He 100% F_{max}	Reg He 100% F_{max}	Reg He 100% F_{max}	Reg He 100% F_{max}	Reg He 100% F_{max}	Reg He 100% F_{max}	Reg He 100% F_{max}		Reg He	Reg He
Vehicle Gr. Wt (est)	400	660	1050	910	1200	1550	1270	1300	1330	1840	1870	1920	1201	1193		1320	2010
No. of Restarts/ Mission	0	0	1-10	1-10	1-10	1-10	1-5	1-5	1-5	1-5	1-5	1-5				1-10	1-10
Positive Expulsion	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	TWO STAGE	TWO STAGE		Yes	Yes
Propellants (Usable)	92.5	230	500	170	440	740	730	750	770	1060	1060	1100				770	1250
Ox.	57	142	307	104	270	455	450	402	475	650	662	672				475	770
Fuel	35.5	88	193	66	170	285	280	288	295	410	418	428				295	480
Actual Gross Wt*	466.7	636.1	1030.2	966.8	1201.2	1536.7	1236.2	1267.8	1303.0	1846.2	1885.6	1926.4	1201.3	1193.3	2380.7	1343.8	2136.5
Actual Δv	2150	4280	6400	2090	4400	6400	8600	8700	8650	8200	8200	8180	8360			8300	8500
Access ΔV	7%	7%	7%	4%	10%	7%	7%	9%	8%	3%	3%	2.9%	4.5%	4.5%	-2%	4%	6.5%

*For 200lb Man Including Suit and ECS

configurations for the same total thrust level are shown on transportation, escape and dual function vehicles. Variation in tank geometry has also been completed on selected configurations to show the effect on vehicle weights and dynamic properties. The entries shown in Table XII were used as a starting point for the conceptual design studies. During the course of the study, as more accurate inputs became available and problem areas were identified, the configurations which displayed distinct disadvantages were eliminated from further refinement.

Selection of an attitude control loop for the Personnel Propulsion Devices represents a task which can be accomplished only through the use of simulation studies by substituting various loops in the system and evaluation of the observed results. In order to enable such evaluation the attitude control systems of all Personnel Propulsion Devices presented in this section have been mechanized to command acceleration since this represents the simplest and more reliable attitude control system and should in fact be used if it produces the required results. It is not possible to predict at this stage of the study, however, whether this mode of control would produce acceptable results with respect to vehicle controllability, propellant consumption and pilot task loading. Information is therefore provided in Section VIII in terms of block diagrams and associated hardware characteristics to enable conversion of the attitude control systems to rate command system, or a rate command system with position hold features, which in previous Bell studies (Ref. 12) has been shown to significantly simplify the control problem, reduce the instability which human operators tend to develop in acceleration command systems and increase the efficiency of propellant expenditure.

Center of gravity excursion in the horizontal plane was established as a major criterion for configuration screening based on the results of the systems analyses presented earlier in Section II. The vehicle configurations employing two tanks, one oxidizer and one fuel installed adjacent to each other, must necessarily be balanced in the fueled condition since the weight of the oxidizer equals 1.6 times the weight of the fuel. This condition necessitates installation of the lift engine at a distance between the tanks so as to provide equal moments. In the burnout condition however, when the propellants are expended, the tanks are of equal weight (same volume and working pressure) consequently, the overall vehicle is unbalanced. Based upon this consideration, all devices employing two tanks, either cylindrical or spherical, were eliminated from design consideration during the studies reported in this section of the report.

The second major criterion used in the initial configuration screening process was the ability of a configuration to have a large overturning angle without excessive landing gear span. Evaluation of the design data, in light of this criterion, restricted the transportation devices to multiple tank arrangements with the tanks aligned horizontally. Vertical alignment of the tanks and pilot with the thrust vector considerably helps to alleviate the balance problem but elevates the center of gravity location necessitating large spans on the landing gear to provide equivalent overturning capability on landing to configurations with the tanks horizontally aligned.

C. TRANSPORTATION DEVICES

Transportation devices are classed into two distinct categories: those which are carried by the man and those which carry the man (vehicles or platforms). The transition from the former to the latter should take place at that point where the dynamic load imposed on the man at touchdown exceeds his ability to support or balance it. In reality, however, this is not represented by a point but rather by a wide range of values caused by the uncertainty of the residual touchdown velocity which is a function of the skill of the operator and the characteristics of the stabilization and control and propulsion systems. Figure 86 presents the envelope of permissible total system load (comprised of the astronaut, life support and propulsion system) versus touchdown velocity. The area below the curve defines the envelope of allowable combinations of system load and residual touchdown velocities for back pack configurations. The boundary line was established by equaling the energy level accrued for a one foot drop on Earth of a 175 pound man. It is believed from operation of the Bell Small Rocket Lift Device that residual velocities less than 2 ft/sec can be attained repeatedly by a trained operator. However a more conservative limit of 4 ft/sec has been applied to lunar operations based on anticipated restrictions of the space suit. Within these constraints the man carried systems offer the most efficient (least weight) configurations. Vehicle or platform configurations are used for one-man transportation devices whose characteristics fall outside this envelope as well as for all two-man transportation devices and escape devices.

Man carried configurations for lunar transportation have been studied in the following forms: (a) waist mounted systems or belt configurations and (b) back mounted systems.

Waist mounted systems have been studied in some detail by Bell Aerosystems for orbital operations (Ref. 6) and by Hamilton Standard (Ref. 7) for application to lunar transportation. The Hamilton Standard OLMS-2 configuration consists of a body mounted unit located about the waist of the man. The unit is powered by four rigidly mounted lift thrusters located at the four extremities of this unit. Two tanks are employed - one on each side and are installed at or about the combined c.g. location of the man and the ECS. The following basic advantages are offered by this configuration:

(1) It permits use of the presently developed environmental control system which is designed for back mounting.

(2) It minimizes the c.g. excursions induced due to propellant consumption; a stringent requirement where rigidly mounted engines are used (see Section II).

Several disadvantages are also apparent in this configuration

(1) The attachment between the man and the propulsion system is less rigid than that provided by the back pack thereby allowing relative motion between the propulsion system and the man and introducing coupling

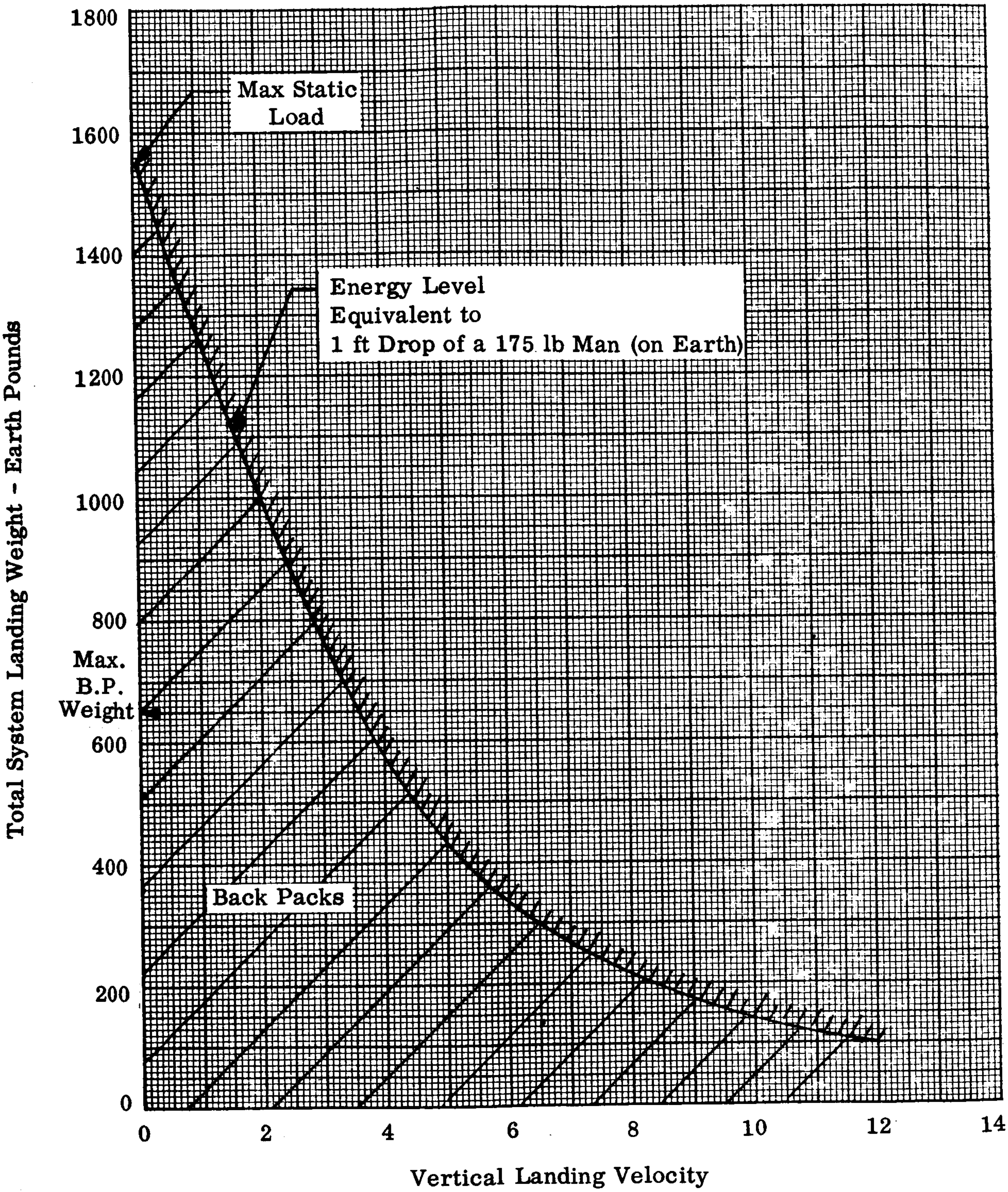


Figure 86. Permissible Weight Envelope for Man Carried Propulsion Systems

uncertainties which may produce destabilizing effects and make control of the device more difficult.

(2) Because of the load transmission characteristics of the device to transmit vertical forces through the pilots boots, it requires a restraint which couples the pilots legs together during flight. This imposes a severe landing restriction since it requires elimination of all residual horizontal velocity to effect a successful landing.

Back pack configurations offer the following significant advantages:

(1) They provide rigid coupling between the suit and the propulsion system, which results in easier control.

(2) They provide freedom to the pilots legs which enables him to absorb vertical loads and to take steps in the direction of the residual horizontal velocity to decelerate his horizontal motion.

Center of gravity excursions in the back pack configurations are larger than those present in waist mounted systems because the consumables are assymetrically loaded. It was shown earlier in Section II, however, that center of gravity excursions can be tolerated in configurations employing gimballed engines, provided that the center of gravity excursion be maintained within the gimbal angle of the thrusters. This condition can be satisfied in back pack arrangements by thorough design studies. In addition, the back pack configurations necessitate repackaging of the life support system on the back and integrating it (physically and maybe functionally) into the propulsion system package or in utilization of a chest mounted life support system.

The back pack configuration has been selected as the model system to be used in the PPD study because it permits greater operational flexibility with respect to landing criteria. That the back pack configurations can be accurately controlled in flight through an open loop (acceleration) system has been demonstrated by the numerous flights conducted by the Small Rocket Lift Device (Ref. 8) whose characteristics are similar to those submitted in this report for simulation studies. In view of these considerations, all further discussions will be restricted to such configurations within their applicable range of operation.

1. One Man

a. Back Pack

Figure 87 presents back pack configurations with a nominal ΔV capability of 2000 and 4000 fps respectively.

The back pack system is compromised of the astronaut, a chest-mounted environmental control system and a back-mounted propulsion system. The pertinent characteristics of these configurations are presented in Table XIII. A weight, center of gravity and inertia summary for the 2000 fps ΔV configuration is presented in Table XIV, c.g. variation as a function of propellant consumption is shown in Figure 88, and a detailed weight statement is shown in Table XV. Similarly, for the 4000 fps ΔV configuration, a weight, center of gravity and inertia summary is shown in Table

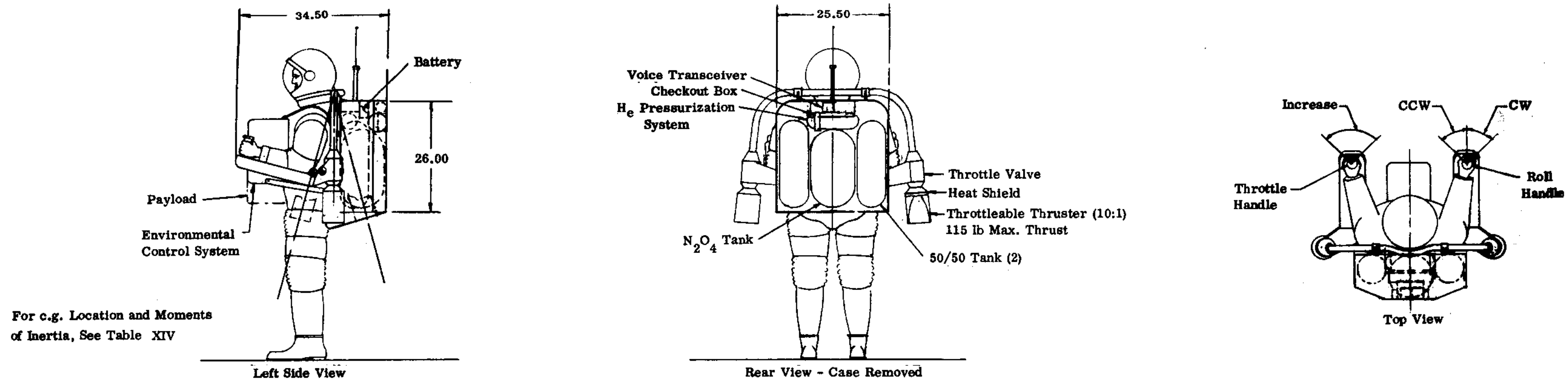


Figure 87a. One Man (Back Pack) Transportation Device
(2000 fps ΔV)

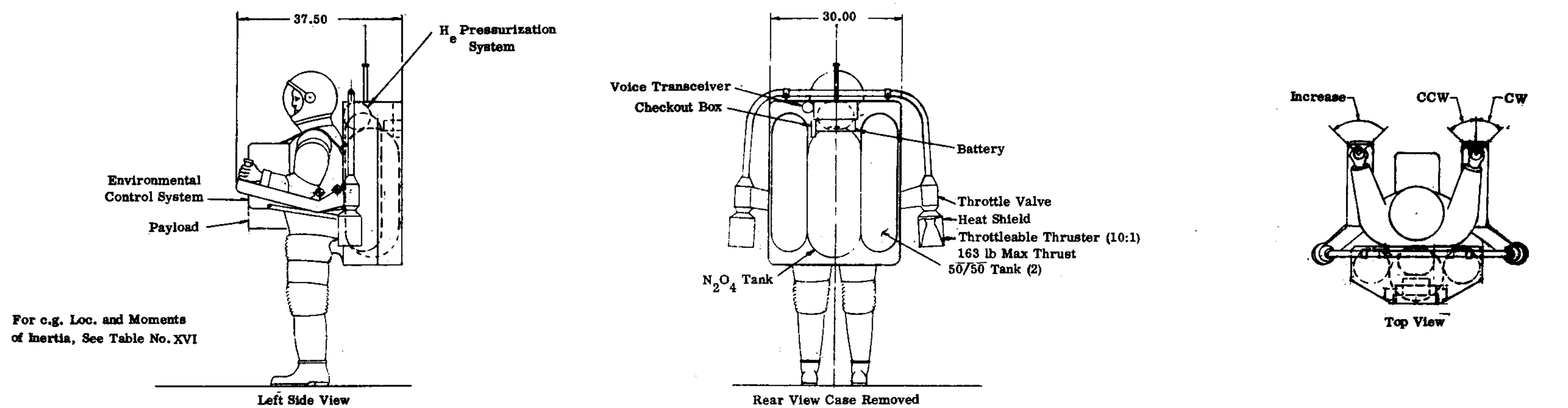


Figure 87b. One Man (Back Pack) Transportation Device
(4000 fps Δ V)

TABLE XIII

BACK PACK CONFIGURATIONS (ONE-MAN TRANSPORTATION)

Figure No.	87a	87b
Nominal ΔV	2000 fps	4000 fps
Gross Weight, Moment of Inertia, c.g. Location & Payload	See Table XIV, Figure 88	See Table XVI, Figure 89
Detailed Weight Statement	See Table XV	See Table XVII
Propulsion System		
Usable Propellants	92.5 lb_e	230.0 lb_e
Thrust level, max ($T/W_i \approx 0.5$)	230.0 lb_f (115 lb each)	326 lb_f (163 lb each)
Tank Configuration	2 fuel, 1 oxidizer	2 fuel, 1 oxidizer
Chamber Pressure	80 psia	80 psia
Tank Pressure	140 psia	140 psia
Positive Expulsion	Teflon bladder; vertical diffuser	Teflon bladder; vertical diffuser
Chamber characteristics	Radiation cooled	Radiation cooled
Throttling Capability	10:1	10:1
Number of Main Propulsion Chambers	2	2
I_{sp}	298 (see insert on Figure 50 for throttle)	299 (see insert on Figure 50 for throttle)
Area Ratio (A_e/A_t)	40	40
Control Mode		
Acceleration in the X direction	Thrust vector	Thrust vector
Acceleration in the Y direction	None provided	None provided
Acceleration in the Z direction	Cumulative throttle control	Cumulative throttle control
Pitch	Cumulative swivel	Cumulative swivel
Yaw	Differential swivel	Differential swivel
Roll	Differential throttle	Differential throttle
Environmental Control System	Chest Pack--4 hr duration	Chest Pack--4 hr duration
Landing Gear		
Equipment		
Voice Communications Transceiver		
Checkout Circuit "Go-No Go"		
Power Supply		
Displays		
Fuel Level (visual and audio)		
Battery Condition		
Controls		
Throttle (rotary handle)		
Arm Controller		
Vehicle Parameters		
Engine Separation Distance (to ℄) d_E	19.5 in.	21.5 in.
Gimbal Angle δ	±20°	±20°
Gimbal Angle Resolution $\Delta\delta$	±1° (Manual)	±1° (Manual)
Gimbal to c.g. Distance d_G	See Figure 88	See Figure 89

TABLE XIV

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA
ONE MAN (BACK PACK) TRANSPORTATION-3 CYLINDRICAL PROPELLANT TANKS

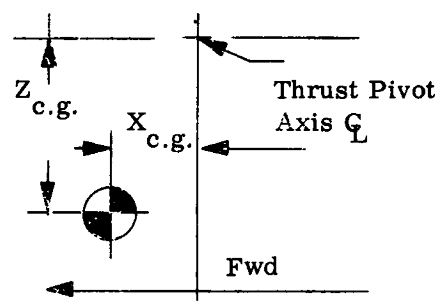
Item	Units	170 lb Man (Incl. Suit)						200 lb Man (Incl. Suit)					
		With Payload			Without Payload			With Payload			Without Payload		
		0%	50%	100%	0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	344.2	391.4	436.7	314.2	361.4	406.7	374.2	421.4	466.7	344.2	391.4	436.7
$X_{c.g.}$ (1)	in.	5.12	3.77	2.71	4.19	2.85	1.82	5.20	3.93	2.92	35	3.09	2.10
$Z_{c.g.}$ (2)	in.	17.74	18.02	17.67	17.04	17,44	17.13	18.05	18.28	17.92	17.45	17.77	17.44
I_{xx}	Slug Ft^2	18.16	18.66	19.31	17.71	18.24	18.86	20.40	20.88	21.64	19.98	20.48	21.12
I_{yy}	Slug Ft^2	20.19	21.53	22.87	19.09	20.27	21.33	22.43	23.79	25.10	21.38	22.58	23.71
I_{zz}	Slug Ft^2	7.04	8.51	9.81	6.28	7.57	8.70	7.26	8.76	10.10	6.52	7.84	9.03

Notes:

(1) $X_{c.g.}$ is Horizontal Distance Between c.g. & Pivot ℄, Inches.
$X_{c.g.}$ is Always Fwd of ℄ Thrust Pivot Axis

(2) $Z_{c.g.}$ is Vertical Distance Between c.g. & Pivot ℄, Inches.
$Z_{c.g.}$ is Always Below ℄ Thrust Pivot Axis.

3 XZ Plane is Plane of Symmetry.



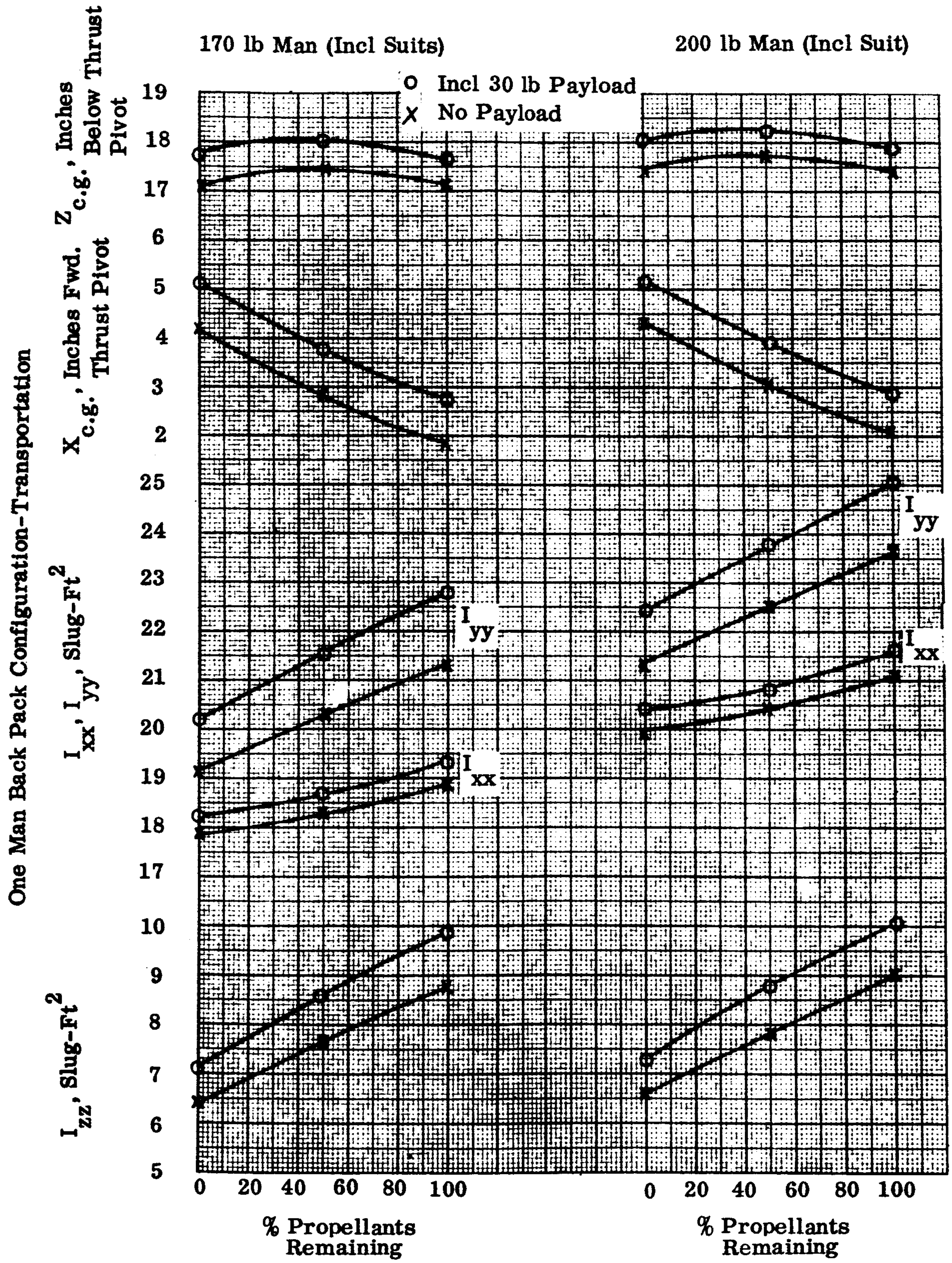


Figure 88. Moments of Inertia and c.g. Variation with Propellants Remaining

TABLE XV

WEIGHT STATEMENT

Items	Weight (Lb)	Items	Weight (Lb)
Back Pack Assy	(78.2)	Payload	(30.0)
Structure	16.0		
Seat	2.0		
Shoulder Strap	0.5	Total Weight Back Pack	172.4
Hip Strap	0.5	Man Weight	170.0
Transceiver (LOS) and Antenna	4.0	Extravehicular Suit	30.0
Fuel and Battery Display	3.5		
Audio Signal for Fuel Indication	0.8	Empty	372.4
Antenna	0.4	Residuals	
Cabling and Plugs	5.0	N_2O_4 (2%)	1.1
Connection to Chest Pack	0.2	50/50 (2%)	0.7
Power Supply	14.7		
Checkout Box "Go No-Go"	2.0	Burnout	374.2
Insulation - Back Pack	6.0	Usables	
N_2O_4 Tank Assy	4.5		
50/50 Tank Assy	6.7	N_2O_4	57.0
Components N_2O_4, 50/50 and He	7.5	50/50	35.5
He Press. Tank incl. He	2.0		
Plumbing N_2O_4, 50/50 and He	1.9	Gross Weight	466.7
Thruster Assembly	(19.2)		
Thrusters (2)	7.6		
Throttle Valves (2)	3.4		
Pivot Tube Assy	3.0		
Control Arm	4.0		
Throttle Handles	1.2		
Chest Pack - ECS	(45.0)		

TABLE XVI

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA
ONE MAN (BACK PACK) TRANSPORTATION - 3 CYLINDRICAL PROPELLANT TANKS

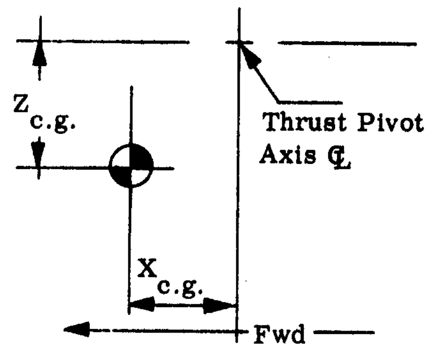
Item	Units	170 lb Man (Including Suit)						200 lb. Man (Including Suit)					
		With Payload Propellants			Without Payload Propellants			With Payload Propellants			Without Payload Propellants		
		0%	50%	100%	0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	376.1	493.4	606.1	346.1	463.4	576.1	406.1	523.0	636.1	376.2	493.5	606.1
$X_{c.g.}$ (1)	in.	7.28	4.56	2.86	6.30	3.65	2.05	7.47	4.86	3.19	6.58	4.02	2.43
$Z_{c.g.}$ (2)	in.	22.07	23.23	21.94	21.37	22.79	21.52	22.41	23.43	22.16	21.80	23.03	21.77
I_{xx}	Slug Ft^2	21.75	24.12	27.37	21.23	23.72	26.84	24.02	26.33	29.64	23.54	25.95	29.15
I_{yy}	Slug Ft^2	24.74	28.64	32.46	23.34	26.91	30.28	27.06	31.03	35.05	25.73	29.38	32.96
I_{zz}	Slug Ft^2	10.62	14.34	17.33	9.63	12.90	15.56	10.87	14.72	17.84	9.92	13.34	16.15

Notes:

(1) $X_{c.g.}$ is Horizontal Distance Between c.g. and Pivot ℄, Inches
$X_{c.g.}$ is Always Fwd of ℄ Thrust Pivot Axis

(2) $Z_{c.g.}$ is Vertical Distance Between c.g. and Pivot ℄, Inches
$Z_{c.g.}$ is Always Below ℄ Thrust Pivot Axis

3. XZ Plane is Plane of Symmetry



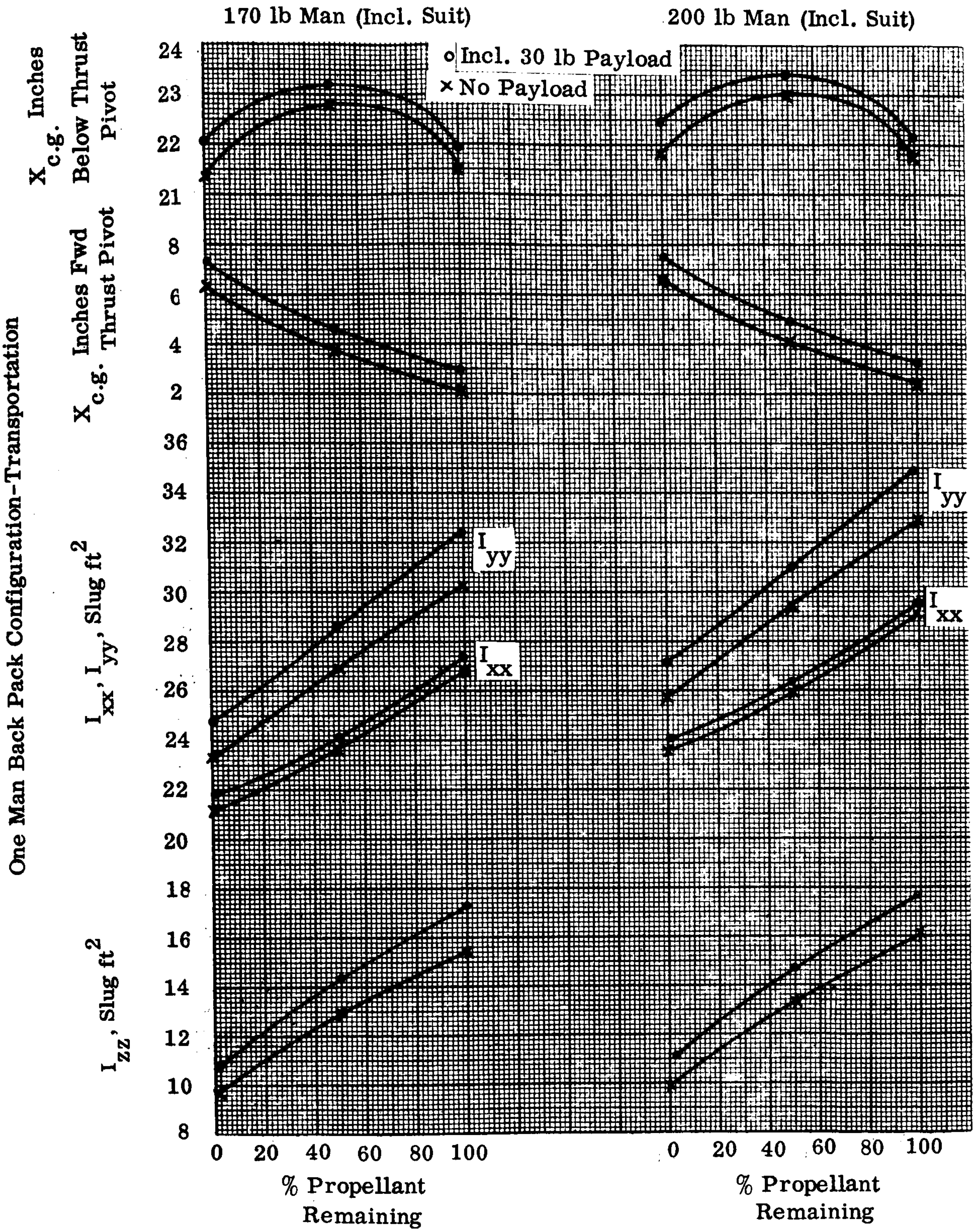


Figure 89. Moments of Inertia and c.g. Variation with Propellants Remaining

TABLE XVII

WEIGHT STATEMENT

Items	Weight (Lb)
Back Pack Assembly	
Structure	22.7
Seat	2.0
Shoulder Strap	0.5
Hip Strap	0.5
Transceiver (LOS) and Antenna	4.0
Fuel and Battery Display and Audio Signal for Fueling	4.3
Antenna	0.4
Cabling and Plugs	5.0
Connection to Chest Pack	0.2
Power Supply	14.7
Checkout Box	2.0
Insulation - Back Pack	9.4
N_2O_4 Tank Assy	8.1
50/50 Tank Assy (2)	10.9
Components N_2O_4, 50/50 and He	7.5
He Press. Tank incl. He	7.0
Plumbing N_2O_4, 50/50 and He	2.6
Thrusters (2)	12.0
Throttle Valves (2)	3.4
Pivot Tube Assy	3.5
Control Arm	4.6
Throttle	1.2
Total Back Pack Assy	126.5

Items	Weight (Lb)
Chest Pack ECS	45.0
Payload	30.0
Total Weight Empty	201.5
Total Weight - Back Pack	201.5
Man Weight	170.0
Extravehicular Suit	30.0
Empty	401.5
Residuals	
N_2O_4 (2%)	2.8
50/50 (2%)	1.8
Burnout	406.1
Usables	
N_2O_4	142.0
50/50	88.0
Gross Weight	636.2

XVI, the c.g. and inertia variation as a function of propellant consumption is shown in Figure 89 , and the detailed weight statement is presented in Table XVII. The ΔV capability of these systems represents only a nominal value, established on the basis of the heaviest combination of astronaut and payload (200 lb astronaut and 30 lb payload) and an assumed throttle setting extrapolated for the entire mission. Insofar as the throttle setting may affect the Δ V capability as much as 15% (see insert on Figure 50), it is highly recommended that the relationship between throttle setting and performance be implemented on the simulation studies.

The back pack propulsion systems employ three propellant tanks to maintain symmetry and alleviate center of gravity excursions in the X Y plane. Four spherical tanks were also considered but resulted in an increase of the center of gravity travel in the X Z plane due to inefficient space utilization within the back pack.

Two throttleable chambers provide thrust for translation and attitude control. The chambers are mounted on a tubular structure which is pivoted by two swivel fittings on the upper end of the back pack structure. Deflection of the chambers for translation and control is manually provided by the astronaut through application of force on the arm rest structure. The chambers are permitted to gimbal $\pm 20^\circ$ in the X Z plane. This action provides thrust vector control for acceleration and attitude maneuvers in the pitch plane. Yaw control is attainable by differential swiveling of the main chambers. Very small gimbal angles (in the order of 2°) are required for yaw control because these configurations have a low moment of inertia about the Z axis.

A segment of the tubular thrust structure located between the swivel fittings is replaced by a torsion bar. This bar permits differential deflection of the thrusters when a moment is applied between the arm rests (i.e. yaw right, is commanded by pushing down on the left arm rest and pulling up on the right arm rest). When the applied moment is relieved, the thrusters will return to the null position by the restoring moment of the torsion bar.

Execution of roll commands by kinesthetic means , that is by displacing his body extremities so as to move the center of gravity and thereby produce a moment in the desired direction based on sensory experience, when the operator is wearing a full pressure suit is questionable at least for the presently developed extravehicular suits. In order to make the propulsion devices independent of the suit characteristics, roll commands are implemented by employing the principle of differential throttling. The operator can command thrust level by rotating the left handle of his controller. This action provides increase or decrease in thrust level in both thrusters in a coordinated manner. The roll command handle, superimposes on the commanded thrust an increment of thrust to one thruster while it removes an equal thrust increment from the opposite thruster.

A chest-mounted environmental control system on all back pack propulsion device configurations provides life support media for a four-hour duration. This system is a repackaged version of the Apollo ECS system used on all vehicle and platform-type configurations. The estimated weight of this package is approximately 45 pounds.

The major advantage of the back pack concept in propulsion systems is the elimination of requirements for a landing gear. The back pack arrangement permits the astronaut to land on one of the most sophisticated landing systems that can be used on any vehicle (his legs) without the associated weight penalty.

Point to point communications on the lunar surface appears feasible for short distances. The line of sight between two objects protruding 10 ft from the surface of the moon, assuring a perfectly smooth but curved surface, is approximately 4 statute miles. Beyond this point either the transmitter or the receiver must be elevated to enable communication between the two stations.

Depending on the mission flight trajectory, even the smallest back pack has a one-way range which exceeds the capability of the ground line-of-sight communication systems. Consequently the communication system required for the transportation devices uses an indirect line of communications (via orbiting spacecraft). The assumption has been made here that an spacecraft is parked in lunar orbit and that communication between the PPD and the lunar base can be made using the orbiting spacecraft as a relay station. The indirect line system, although resulting in a greater weight penalty to the system, does provide continuous communication with the lunar base independent of range and lunar terrain obstructions; however, communications are restricted to the time interval when the orbiting spacecraft is within LOS of both the transportation device and the lunar base.

An automatic checkout circuit is incorporated on all devices to monitor the readiness of the system prior to mission initiation. Insofar as no extensive maintenance is anticipated to be conducted on the moon, only a "Go-No Go" indicator has been provided to ensure the system operational status.

Back pack propulsion systems are intended for use in the proximity of the lunar base and for exploration of areas inaccessible by other vehicles. Consequently no navigation or guidance equipment have been provided on these configurations.

The low power requirements of the Personnel Propulsion Devices coupled with the short period of operation dictate the use of batteries as the most efficient means of electrical power storage. A 14.7 pound power supply consisting of a silver zinc battery and pressurized container provides continuous power for voice communications and displays and intermittent power on demand for operation of the throttle valve. The specific weight of silver zinc batteries for space applications is approximately 40 watt-hours per pound based on a two-hour discharge rate. The power supply includes 100% overload reserve.

The displays provided on the back pack configurations include a battery condition dial indicator incorporated on the roll control handle. The fuel level indicator is installed on the throttle handle. In conjunction with the fuel level display, an intermittent audio signal of 10-second repetition cycle is provided to the astronaut. The signal becomes continuous when the remaining fuel supply reaches the level adequate for 10-second operation at maximum thrust.

b. Vehicle

Figure 90 presents a one-man transportation device with a nominal ΔV capability of 6000 fps. The basic drawing depicts a single engine configuration with four spherical propellant tanks. This arrangement was selected to conduct a comparison study for evaluation of the gross effects of tank geometry and engine arrangement during the configuration refinement period. Without the aid of simulation studies and/or specific mission profiles, the configurations can only be evaluated in terms of their physical characteristics. Table XVIII presents the pertinent characteristics of this configuration. The weight, center of gravity and moment of inertia summary resulting from installation of one, two and four engines to provide the same T/W and from the use of spherical and cylindrical tanks is shown in Table XIXa, b and c. A weight statement for the selected configuration (a single engine with spherical tanks) is presented in Table XX. Variation of inertia with propellant consumption was found to be a linear function; consequently this variation is not presented graphically.

While gross center of gravity excursions due to propellant consumption were eliminated during the first screening operation by proper location of the consumables, center of gravity uncertainties smaller in magnitude may still arise from unequal drainage of propellant from the main tanks due to line impedance. These center of gravity shifts reach a maximum at the burnout condition. The limiting cases are shown in Table XVIII. The center of gravity uncertainties, together with the selected control acceleration requirements, were used to arrive at the thrust level of the attitude control thrusters.

Cylindrical tanks on the component level are obviously heavier than spherical tanks; however, this does not always hold true when the comparison is made on the system level. Therefore vehicles were configured using spherical and cylindrical tanks and weight statements prepared for both. The results of this comparison indicate a 5.6 pound system weight penalty was realized from the net change of spherical to cylindrical tanks. Consequently spherical tanks have been selected for this and subsequent vehicle designs unless space limitations in vehicle arrangement dictate otherwise. The changes in the vehicle moment of inertia brought about by changing the tank configuration may be considered to be insignificant.

One-, two- and four-engine arrangements have also been studied to determine whether gross changes in the physical properties of the vehicle would result. This was accomplished by holding all other parameters constant (unless necessitated by the engine configuration) and determining the change in weight, inertia and dynamic properties of the vehicle for the various engine arrangements. For equal total thrust and area ratio the single engine provides the least weight. Other features are provided by multiple engine configurations; however, these effects can only be evaluated in terms of performance and mission profile. The delivered specific impulse at rated thrust and associated area ratio permitted by envelope restrictions for each vehicle is presented in Table XX. It may be noted that if the trajectory requires engine operation at maximum thrust, the two engine configurations should provide the maximum ΔV capability. However if a large percentage of the trajectory requires less than 50% of maximum thrust the four-engine configuration could provide superior performance by operating with two engines "off" while the remaining engines operate in the more efficient range of thrust level (near rated thrust). In addition the four-engine configuration provides 100% engine redundancy. However without the aid of simulation and specific mission profiles, the engine configuration cannot be selected for optimum performance.

The attitude control system is comprised of six radiation-cooled attitude thrusters fed from the main propulsion tanks. This arrangement was found to result in minimum weight penalty to the vehicle. A detailed tradeoff study which substantiates this selection is shown in Section III of this report.

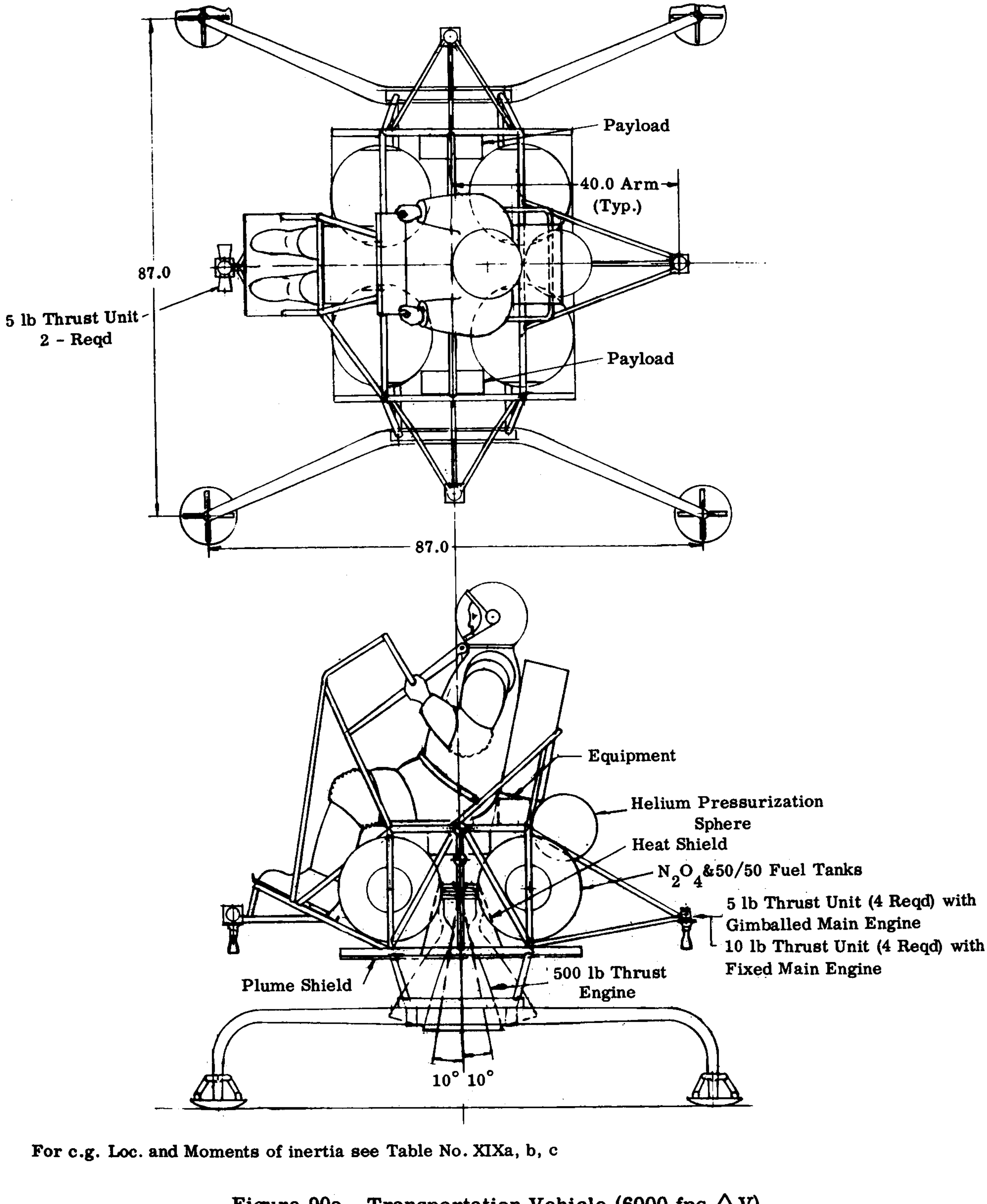
Since this vehicle is not designed for a specific astronaut, but rather for astronauts whose weight ranges from 140 to 170 earth pounds, provisions have been made to permit fore and aft adjustment of the seat to allow the c.g. of the man to coincide with that of the vehicle in the horizontal plane. This simple adjustment permits reduction of the attitude chamber thrust level by elimination of the initial c.g. error.

The vehicle is designed with tubular welded aluminum to form a modified truss which supports the tanks, engine(s), equipment and landing gear. A discussion of the structural arrangement, criteria and landing gear characteristics is presented in Section VI. The equipment compartment is located on the vehicle directly below the astronaut's environmental control system. This compartment is insulated to provide passive environmental control for the electronic equipment.

In keeping with the objective of employing manual systems, all controllers should be mechanized to command acceleration at least in the initial simulation runs, to determine whether this mode of control produces acceptable results. If the results indicate that improvement in the control system characteristics are desirable, then one of the more sophisticated loops, i.e. rate command or positive hold, should be substituted in the vehicle and the changes in controllability and performance monitored until the required results are attained. As will be shown later in Figure 108 of Section VIII, little difference in weight exists between the simplest acceleration system and the improved rate command system with position hold features (approximately 4 lb). However, the pilot task loading variation imposed by each of these systems is vastly different. The time available for control, stabilization, navigation, as well as the displays necessary for the vehicle can only be defined after completion of simulation studies with specified trajectories.

2. Two-Man Devices

Two-man transportation devices for 2000, 4000 and 6000 fps Δ V are shown on Figures 91, 92, and 93, respectively. Pertinent characteristics of these configurations are shown in Table XXI. The weight, center of gravity and moment of inertia summary for the 2000 fps ΔV configuration shown in Figure 91 is presented in Table XXIII. A weight statement for this configuration is shown in Table XXIV. The weight center of gravity and moment of inertia summary for the 4000 fps Δ V configuration, which is shown in Figure 92, is presented in Table XXV and the corresponding weight statement in Table XXVI. In similar manner, the weight center of gravity and inertia table for the 6000 fps ΔV configuration shown in Figure 93 is presented in Table XXVII and the weight statement in Table XXVIII. The main objective here has been



For c.g. Loc. and Moments of inertia see Table No. XIXa, b, c

Figure 90a. Transportation Vehicle (6000 fps ΔV)

For c.g. Loc & Moments of Inertia, see Table XIX

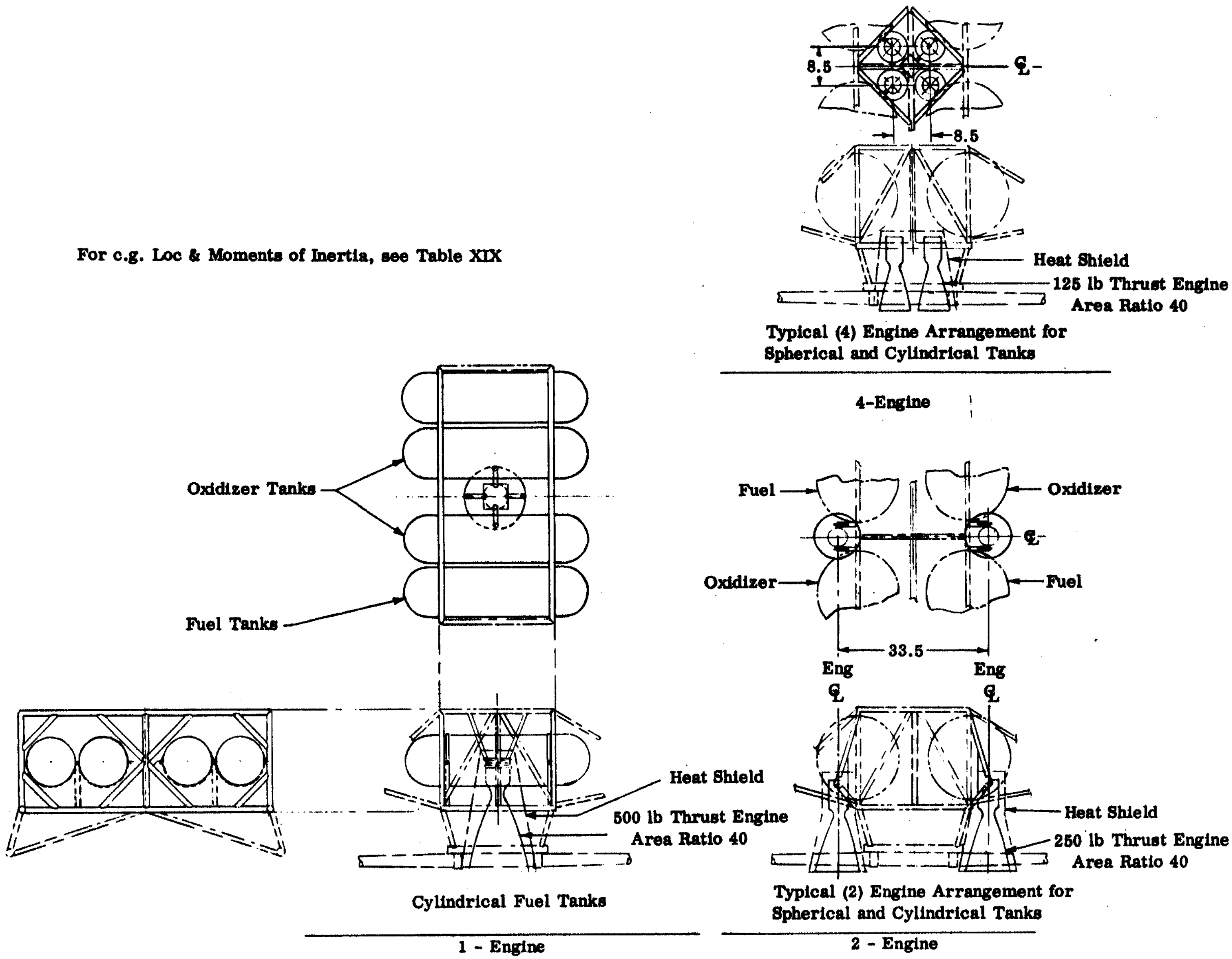


Figure 90b. Alternate Arrangements

TABLE XVIII

ONE MAN TRANSPORTATION DEVICE (VEHICLE)

Figure No.	90		
Nominal Δ V	6000 fps		
Number of Engines in Main Propulsion	1	2	3
Gross Weight, Moments of Inertia, c.g. Location	See Table XIX (a)	XIX (b)	XIX (c)
Detail Weight Statement	See Table XX		
Propulsion System			
Usable Propellants	500 lb_e		
Thrust Level Max. (T/W_i 0.5)	500 lb_f		
Tank Configuration	2 fuel, 2 oxidizer		
Chamber Pressure	80 psia		
Tank Pressure	140 psia		
Positive Expulsion	Teflon Bladder		
Chamber Characteristics	Radiation Cooled		
Throttling Capability	10:1	10:1	20:1
I_{sp}	301*	302.2*	295.5*
Area Ratio (A_e/A_t)--Hover Chambers	40**	60	40**
Control Mode			
Acceleration in the X Direction	Thrust Vector Positioning		
Acceleration in the Y Direction	Thrust Vector Positioning		
Acceleration in the Z Direction	Throttle Control		
Pitch	Unbalanced Moment		
Yaw	6 Chamber Attitude Control		
Roll	System		
Environmental Control System	Apollo Back Pack--4 hr duration		
Landing Gear	Tubular fiber glass 10 fps vertical equivalent; reusable		
Equipment	Group A (See Section V)		
Vehicle Parameters--Throttleable Engines d_E		16.75 in.	4.25 in.
Vehicle Parameters--Gimballed Engines			
Gimbal Angle δ	±5.2°		
Gimbal Angle Resolution $\Delta\delta$	0.15° (for 1°/sec^2 residual accel.)		
Gimbal to c.g. distance d_G	9.0 in.		
Max c.g. Excursions ϵ_x	±0.13 in.		
ϵ_y	±0.13 in.		
ϵ_z	+5.2 in.		
Reaction Control System	N_2O_4/50-50 fed from main tanks		
Commanded Acceleration	10°/sec^2		
Lever Arm P, Y, R	40 in.		
Number of Thrusters	6		
Thruster Size	5 lb with gimballed main engine 10 lb with fixed main engines for pitch & roll 5 lb with fixed main engines for yaw		

*See insert on Figure 50 for I_{sp} degradation due to throttling

**Envelope limitations

TABLE XIX
WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA

(a) One Man Transportation Vehicle - (1) 500 lb Engine (T/W = 0.5)

		170 lb Man (Including Suit)						200 lb Man (Including Suit)					
		(4) Spherical Tanks Propellants			(4) Cylindrical Tanks Propellants			(4) Spherical Tanks Propellants			(4) Cylindrical Tanks Propellants		
Items	Units	0%	50%	100%	0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	500.2	750.2	1000.2	505.8	755.8	1005.8	530.2	780.2	1030.2	535.8	785.8	1035.8
$X_{c.g.}$ ⚠1	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Z_{c.g.}$ ⚠2	in.	47.64	43.17	42.72	47.56	43.66	42.86	48.48	43.91	43.29	48.40	44.38	43.43
I_{xx}	Slug ft^2	58.0	73.0	80.9	59.2	76.6	91.2	60.5	76.5	84.6	61.8	80.0	94.8
I_{yy}	Slug ft^2	62.0	77.0	85.0	61.5	71.5	75.8	64.9	80.9	89.0	64.5	75.3	79.8
I_{zz}	Slug ft^2	34.7	52.2	69.9	35.7	53.5	69.9	35.4	52.8	70.5	36.3	54.1	70.5
ϵ_x	in.	±0.13			0			±0.12			0		
ϵ_y	in.	±0.13			±0.14			0.12			±0.13		

(b) One Man Transportation Vehicle - (2) 250 lb Engines (T/W = 0.5)

Items	Units	0%	50%	100%	0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	509.5	759.5	1009.5	515.1	765.1	1015.1	539.5	789.5	1039.5	545.1	795.1	1045.1
$X_{c.g.}$ ⚠1	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Z_{c.g.}$ ⚠2	in.	47.01	42.81	42.46	46.94	43.29	42.60	47.88	43.55	43.03	47.80	44.02	43.17
I_{xx}	Slug ft^2	59.7	74.2	82.1	60.9	77.8	92.4	62.4	77.8	85.8	63.6	81.3	96.0
I_{yy}	Slug ft^2	66.0	80.5	88.4	65.6	75.1	79.2	69.0	84.5	92.5	68.6	78.9	83.3
I_{zz}	Slug ft^2	37.0	54.4	72.2	37.9	55.7	72.1	37.6	55.1	72.8	38.6	56.4	72.7
ϵ_x	in.	±0.13			0			±0.12			0		
ϵ_y	in.	±0.13			±0.14			±0.12			±0.13		

(c) One Man Transportation Vehicle - (4) 125 lb Engines (T/W = 0.5)

Items	Units	0%	50%	100%	0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	509.62	759.62	1009.62	515.22	765.22	1015.2	539.62	789.62	1039.62	545.22	795.22	1045.22
$X_{c.g.}$ ⚠1	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Z_{c.g.}$ ⚠2	in.	46.77	42.65	42.34	46.71	43.14	42.48	47.66	43.41	42.92	47.59	43.87	43.05
I_{xx}	Slug ft^2	60.8	75.1	83.0	62.0	78.8	93.2	63.5	78.7	86.7	64.8	82.3	96.9
I_{yy}	Slug ft^2	64.8	79.1	87.0	64.4	73.7	77.9	67.9	83.1	91.1	67.5	77.6	81.9
I_{zz}	Slug ft^2	35.0	52.4	70.1	35.9	53.7	70.1	35.6	53.0	70.8	36.5	54.3	70.7
ϵ_x	in.	±0.13			0			±0.12			0		
ϵ_y	in.	±0.13			±0.14			±0.12			±0.13		

Notes:

⚠1 $X_{c.g.}$ is Horizontal Distance Fwd (+), Aft (-) From ℄ of Vertical Thrust

⚠2 $Z_{c.g.}$ is Vertical Distance Above Ground Line.

3 X Z Plane is Plane of Symmetry.

4 Propellant c.g.'s are Calculated for Propellants in a Static Condition

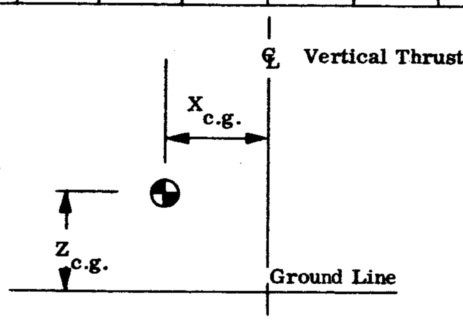
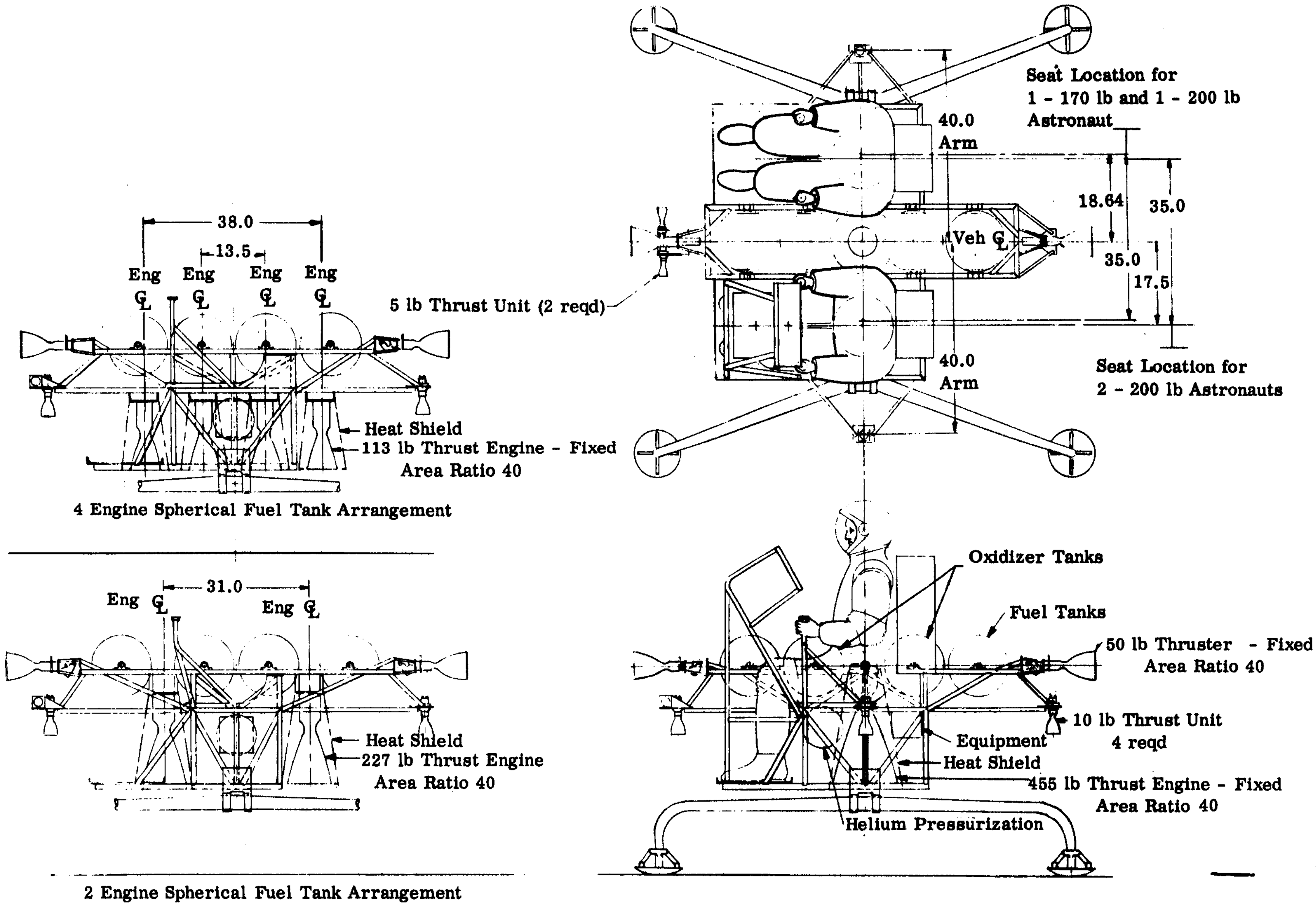


TABLE XX

WEIGHT STATEMENT

Items	Weight (Lb)	Items	Weight (Lb)
Structure	(55.1)	Attitude Control	(13.4)
Tubular Framework	30.4	Thrusters - (2) 5 lb	3.0
Honeycomb - Plume Protection	8.0	Thrusters - (4) 10 lb	6.4
Supports - Propellant Tanks	3.2	Plumbing	4.0
Supports - Pressurization Tank	0.8	Heatshield	-
Supports - Thrust Chamber	-	Guidance (Manual) System and Installation	8.5
Supports - Attitude Control Thrusters	3.4	Equipment (Group A) (See Section V)	41.2
Supports - Electronic and Communication	2.0	Furnishings	(3.5)
Supports - Guidance	1.4	Seat Assembly	3.0
Supports - Foot and Control	5.9	Seat Belt	0.5
Supports - Hand	-	Shoulder Strap	-
Supports - Miscellaneous	-		
Propulsion System	(55.2)	Landing Gear Installation	21.6
N_2O_4 Tank Assy (2) Spherical ⚠4	17.6	Crew (incl Suit) ⚠1 (1) Man	200.0
50/50 (N_2H_4 - UDMH) Tank Assy (2) Spherical ⚠4	17.6	Back Pack - Life Support System (1)	45.0
Plumbing and Valves	8.0	Residuals - Ox., N_2O_4 (2%)	6.1
Insulation (Tank)	12.0	Residuals - Fuel, 50/50 (N_2H_4-UDMH)(2%)	3.9
Miscellaneous	-	Payload	30.0
Pressurization System	(16.4)	Burnout Weight	530.2
Tank (incl. Gas)	11.9	Usables - Ox., N_2O_4 ⚠2	307.0
Plumbing and Valves	4.0	Usables - Fuel, 50/50 (N_2H_4-UDMH) ⚠3	193.0
Insulation (Tank)	0.5	Gross Weight	1030.2
Engine Installation	(30.3)		
Thrust Chamber (1) 500 lb ⚠5	15.0		
Biprop Valve	2.3		
Heatshield	1.0		
Gimballed System (incl. Mt.)	12.0		

⚠1 See Weight, Center of Gravity and Moment of Inertia Summary Sheets for Variations in the Weight of the Man.
⚠2 Includes 18.4 Pounds for Attitude Control.
⚠3 Includes 11.5 Pounds for Attitude Control.
⚠4 See Weight and Balance Summary Sheets for Variations in Tank Weights.
⚠5 See Weight and Balance Summary Sheets for Variations in Thrust Chamber Weights.



For c.g. Loc & Moments of Inertia, See Table XXIII

Figure 91. 2 - Man Transportation Vehicle (2000 fps ΔV)

TABLE XXI

TWO-MAN TRANSPORTATION DEVICES

Figure No.	91	92	93
Nominal Δ V	2000 fps	4000 fps	6000 fps
Number of Engines in Main Propulsion Vertical Horizontal	1 2 4 2 2 2	1 2 4	1 2 4
Gross Weight, Moments of Inertia, c.g. Location	See Table XXIII	See Table XXV	See Table XXVII
Detailed Weight Statement	See Table XXIV	See Table XXVI	See Table XXVIII
Propulsion System			
Usable Propellants (including reaction propellants)	170 lb_e	440 lb_e	740 lb_e
Thrust Level max. (T/W_i 0.5)	455 lb_f	600 lb_f	775 lb_f
Tank Configuration	2 fuel, 2 oxidizer	2 fuel, 2 oxidizer	2 fuel, 2 oxidizer
Chamber Pressure	80 psia	80 psia	80 psia
Tank Pressure	140 psia	140 psia	140 psia
Positive Expulsion	Teflon bladders	Teflon bladders	Teflon bladders
Chamber Characteristics			
Throttling Capability	10:1 10:1 20:1	10:1 10:1 20:1	10:1 10:1 20:1
I_{sp}	300.6* 297.7* 294.5*	301.7* 295.7* *	305.2* 300.0* 297.0*
Area Ratio (A_e/A_t) - Hover Chambers	40 40 40	40 40 40	40 40 40
Control Mode			
Acceleration in the X Direction	Horizontal thrusters	Thrust vector positioning	Thrust vector positioning
Acceleration in the Y Direction	Yaw & use horizontal thrusters	Thrust vector positioning	Thrust vector positioning
Acceleration in the Z Direction	Throttle control	Throttle control	Throttle control
Pitch	Unbalanced moment	Unbalanced moment	Unbalanced moment
Yaw Roll	6 Chamber attitude control system	6 Chamber attitude control system	6 Chamber attitude control system
Environmental Control System	Apollo back pack - 4 hr duration	Apollo back pack - 4 hr duration	Apollo back pack - 4 hr duration
Landing Gear	Tubular Fiberglass 10 fps vertical equiv. reusable	Tubular Fiberglass 10 fps vertical equiv. reusable	Tubular fiberglass 10 fps vertical equiv. reusable
Equipment	Group A (see Section V)	Group A (see Section V)	Group A (see Section V)
Vehicle Parameters - Throttleable Engines d_E	15.5 in. 19.0 in.	9.0 in. 7.0 in.	9.5 in. 7.5 in.
Vehicle Parameters - Gimballed Engines Gimbal Angle δ Gimbal Angle Resolution $\Delta\delta$ Gimbal to c.g. Distance d_G	Fixed Engines	Fixed Engines	Fixed Engines
Max. c.g. Excursions ϵ_x ϵ_y	±0.04 in. 0.0 in. ±0.55 in.	±0.09 in. ±0.09 in. 6.22 in.	±0.15 in. ±0.15 in. 8.0 in.
Reaction Control System	N_2O_4/50-50 fed from main tanks	N_2O_4/50-50 fed from main tanks	N_2O_4/50-50 fed from main tanks
Commanded Acceleration	10°/sec^2	10°/sec^2	10°/sec^2
Lever Arm P, Y, R	40 in.	40 in.	40 in.
Number of Thrusters	6	6	6
Thruster Size	10 lb_f (pitch & yaw)	10 lb_f (pitch & roll)	10 lb_f (pitch and roll)
Thruster Size	5 lb_f (yaw)	5 lb_f (yaw)	5 lb_f (yaw)

* See insert on Figure 50 for I_{sp} degradation due to throttling

TABLE XXIII

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA
TWO MAN TRANSPORTATION VEHICLE - 4 SPHERICAL PROPELLANT TANKS
(1) ENGINE-THRUST LEVEL = 500 LB (T/W = 0.5)

Items	Units	(2) 170 lb Men (Incl. Suits)			(2) 200 lb Men (Incl. Suits)			(1) 170 lb & 200 lb Man (Incl. Suits)		
		Propellants Remaining			Propellants Remaining			Propellants Remaining		
		0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	638.8	723.8	808.3	698.8	783.8	868.8	668.8	753.8	838.8
$X_{c.g.}$ (1)	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Y_{c.g.}$ (4)	in.	0.00	0.00	0.00	0.00	0.00	0.00	-0.80[5]	-0.70[5]	-0.63[5]
$Z_{c.g.}$ (2)	in.	40.57	40.03	40.13	40.97	40.42	40.48	40.78	40.23	40.31
I_{xx}	Slug Ft^2	66.9	67.1	67.1	73.5	73.8	73.8	70.1	70.4	70.4
I_{yy}	Slug Ft^2	51.8	57.6	63.2	55.6	61.6	67.4	53.9	59.7	65.4
I_{zz}	Slug Ft^2	62.2	67.8	73.5	67.8	73.6	79.4	65.1	70.7	76.4
ϵ_x	in.	±0.04			±0.04			±0.04		
ϵ_y	in.	0			0			0		

Notes:
(1) $X_{c.g.}$ is the Horizontal Distance Fwd (+) or Aft (-) From the ℄ of the Vertical Thrust Engine.
(2) $X_{c.g.}$ is the Vertical Distance Above Ground Line.
3 XZ Plane is Plane of Symmetry.
4 $Y_{c.g.}$ is the Lateral Distance L.H. (-) or R.H. (+) From XZ Plane of Symmetry.
5 Indicates c.g. Travel if No Seat Adjustment is Made.

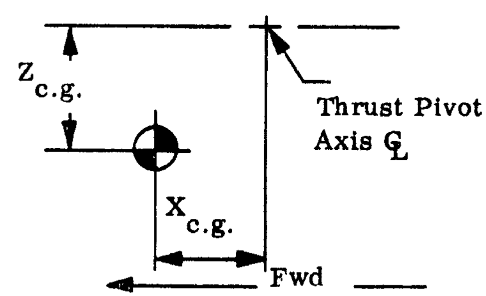


TABLE XXIV

WEIGHT STATEMENT

Items	Weight (Lb)	Items	Weight (Lb)
Structure	(48.0)	Attitude Control	(13.9)
Tubular Framework	25.2	Thrusters - (4) at 10 lb	6.4
Honeycomb - Plume Protection	9.5	- (2) at 5 lb	3.0
Supports - Propellant Tanks	2.0	Plumbing	4.5
- Pressurization Tank	0.5	Heatshield	-
- Thrust Chamber	1.4	Guidance (Manual) System and Installation	8.5
- Attitude Control Thrusters	4.5		
- Electronic and Communication	2.0	Equipment (Group A)	41.2
- Guidance	1.4	Furnishings	(7.0)
- Foot (incl. in Plume Protection	-	Seat Assembly (2)	6.0
- Hand	1.5	Seat Belt (2)	1.0
- Miscellaneous	-	Shoulder Strap	-
Propulsion System	(29.4)	Landing Gear Installation	22.8
N_2O_4 Tank Assy	8.2		
50/50 (N_2H_4 - UDMH) Tank Assy	8.2	Crew (incl. Suit) △1 (2)	400.0
Plumbing and Valves	9.5	Back Pack - Life Support System (2)	90.0
Insulation (Tank)	3.5		
Miscellaneous	-	Residuals - Ox., N_2O_4 (2%)	2.1
		- Fuel, 50/50 (N_2H_4-UDMH) (2%)	1.3
Pressurization System	(8.2)		
Tank (incl. Gas)	3.2	Payload	-
Plumbing and Valves	4.5		
Insulation (Tank)	0.5	Burnout Weight	698.8
		Usables - Ox., N_2O_4 △2	104.0
Engine Installation	(26.4)	- Fuel, 50/50 (N_2H_4-UDMH) △3	66.0
Thrust Chamber (1) at 500 lb	14.5		
Biprop Valve	2.3	Gross Weight	868.8
Heatshield	3.0		
Gimballed System	-		
Horizontal Thrusters - (2) at 50 lb	6.6		

△1 See Weight, Center of Gravity and Moment of Inertia Summary Sheets for Variations in the Weight of the Man.
△2 Includes 18.4 Pounds for Attitude Control.
△3 Includes 11.5 Pounds for Attitude Control.

TABLE XXV

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA
(2) MAN TRANSPORTATION VEHICLE - (4) SPHERICAL PROPELLANT TANKS
(1) ENGINE THRUST LEVEL AT 600 LB (T/W = 0.5)

Items	Units	(2) 170 lb Men (Incl. Suit)			(2) 200 lb Men (Incl. Suit)			(1) 170 lb & (1) 200 lb Man (Incl. Suit)		
		Propellants Remaining			Propellants Remaining			Propellants Remaining		
		0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	701.2	921.2	1141.2	761.2	981.2	1201.2	731.2	951.2	1171.2
$X_{c.g.}$ (1)	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Y_{c.g.}$ (4)	in.	0.00	0.00	0.00	0.00	0.00	0.00	-0.56[5]	-0.43[5]	-0.35[5]
$Z_{c.g.}$ (2)	in.	47.17	42.54	40.95	47.87	43.36	41.70	47.53	42.96	41.34
I_{xx}	Slug Ft^2	81.7	105.3	116.8	87.6	112.5	124.7	84.6	108.9	120.7
I_{yy}	Slug Ft^2	66.1	89.7	101.3	70.3	95.2	107.3	68.2	92.5	104.3
I_{zz}	Slug Ft^2	64.8	84.0	103.4	68.7	87.8	107.2	66.7	85.9	105.3
ϵ_x	in.	±0.09			±0.08			±0.08		
ϵ_y	in.	±0.09			±0.08			±0.08		

Notes:
1. $X_{c.g.}$ is Horizontal Distance Fwd (+), Aft (-) From ℄ of Vertical Thrust Engine.
2. $Z_{c.g.}$ is Vertical Distance Above Ground Line.
3. XZ Plane is Plane of Symmetry.
4. $Y_{c.g.}$ is Lateral Distance L.H. (-), R.H. (+) From XZ Plane of Symmetry.
5. c.g. Travel if No Seat Adjustment is Made.

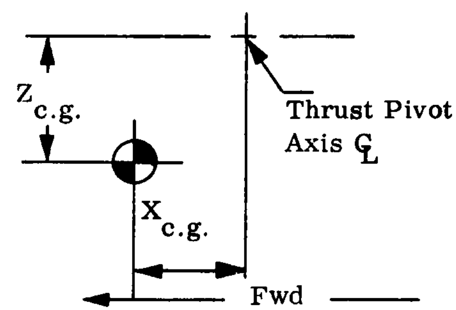


TABLE XXVI

WEIGHT STATEMENT

Items	Weight (Lb)
Structure	(72.1)
Tubular Framework	35.4
Honeycomb - Plume Protection	7.2
Supports - Propellant Tanks	3.2
- Pressurization Tank	0.8
- Thrust Chamber	1.4
- Attitude Control Thrusters	4.1
- Electronic and Communication	2.0
- Guidance	1.4
- Foot	9.6
- Hand	2.6
- Miscellaneous (Life Supt. Sys)	4.4
Propulsion System	(53.2)
N_2O_4 Tank Assy (2)	16.8
50/50 (N_2H_4-UDMH) Tank Assy (2)	16.8
Plumbing and Valves	8.0
Insulation (Tank)	11.6
Miscellaneous	-
Pressurization System	(18.3)
Tank (incl. Gas)	11.3
Plumbing and Valves	6.0
Insulation (Tank)	1.0
Engine Installation	(20.7)
Thrust Chamber (1) at 600 lb	16.0
Biprop Valve	2.6
Heatshield	2.1
Gimballed System	-

Items	Weight (Lb)
Attitude Control	(13.4)
Thrusters - (4) at 10 lb	6.4
- (2) at 5 lb	3.0
Plumbing	4.0
Guidance (Manual) System and Installation	8.5
Equipment (Group A)	41.2
Furnishings	(7.0)
Seat Assembly (2)	6.0
Seat Belt (2)	1.0
Shoulder Strap	-
Landing Gear Installation	28.0
Crew (incl. Suit) △1 (2)	400.0
Back Pack - Life Support System (2)	90.0
Residuals - Ox., N_2O_4 (2%)	5.4
- Fuel, 50/50 (N_2H_4-UDMH)(2%)	3.4
Payload	-
Burnout Weight	761.2
Usables - Ox., N_2O_4 △2	270.0
- Fuel, 50/50 (N_2H_4-UDMH) △3	170.0
Gross Weight	1201.2

△1 See Weight, Center of Gravity and Moment of Inertia Summary Sheets for Variations in the Weight of the Man.
△2 Includes 18.4 Pounds for Attitude Control.
△3 Includes 11.5 Pounds for Attitude Control.

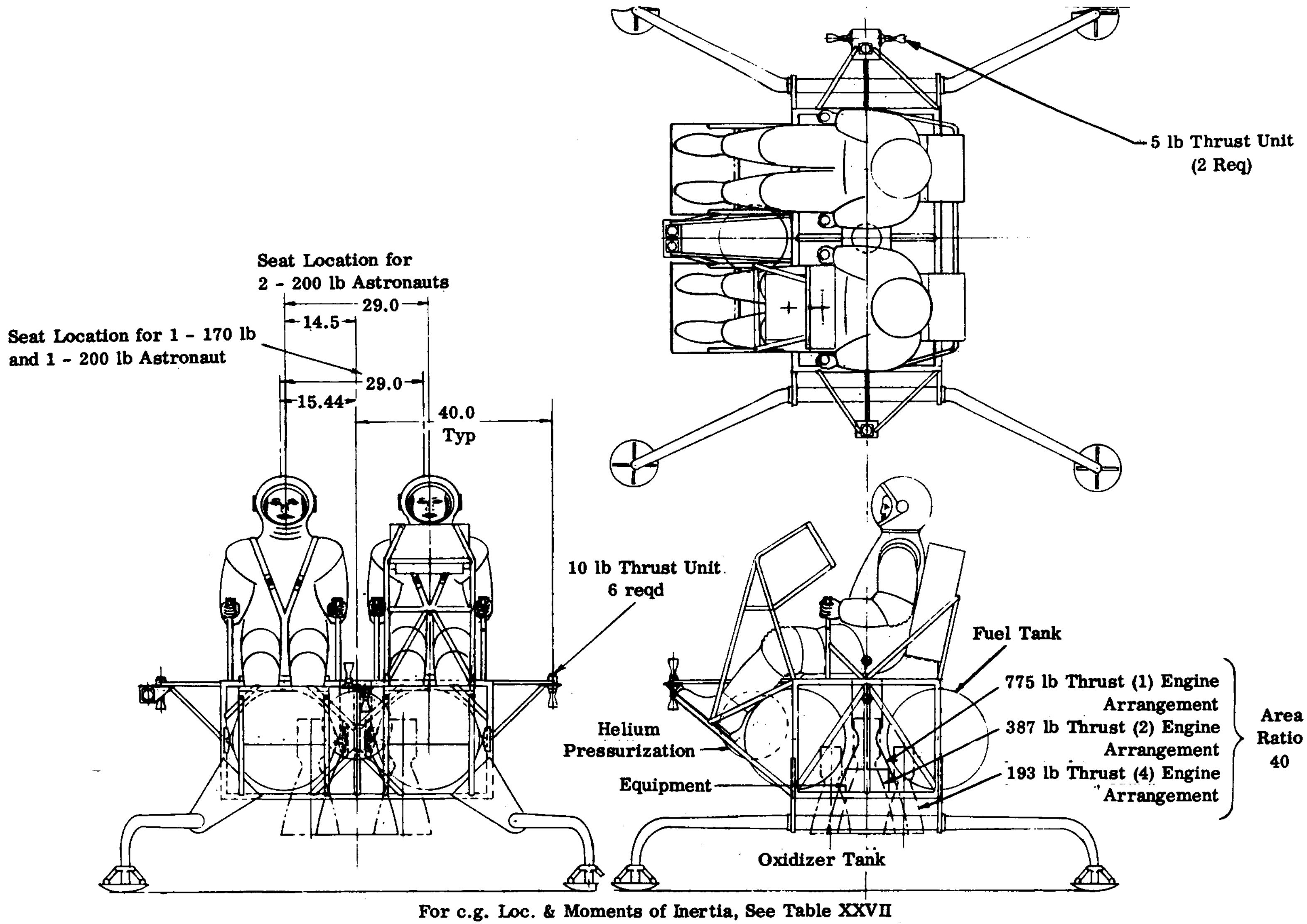


Figure 93. 2-Man Transportation Vehicle (6000 fps Δ V)

TABLE XXVII

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA
(2) MAN TRANSPORTATION VEHICLE - (4) SPHERICAL PROPELLANT TANKS
(1) ENGINE THRUST LEVEL AT 775 LB (T/W = 0.5)

Items	Units	(1) 170 lb Men (Incl. Suit) Propellants Remaining			(2) 200 lb Men (Incl. Suit) Propellants Remaining			(1) 170 lb & (1) 200 lb Man (Incl. Suit) Propellants Remaining		
		0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	738.7	1108.7	1478.7	798.7	1168.7	1538.7	768.7	1138.7	1508.7
$X_{c.g.}$ (1)	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Y_{c.g.}$ (4)	in.	0.00	0.00	0.00	0.00	0.00	0.00	-0.57 [5]	-0.38 [5]	-0.29 [5]
$Z_{c.g.}$ 2	in.	47.39	40.77	39.42	48.15	41.62	40.12	47.79	41.21	39.78
I_{xx}	Slug Ft^2	96.7	135.9	153.7	103.2	144.6	163.1	99.9	140.2	158.4
I_{yy}	Slug Ft^2	78.4	117.6	135.5	82.9	124.4	142.8	80.7	121.0	139.2
I_{zz}	Slug Ft^2	78.5	114.1	150.2	82.6	118.2	154.3	80.5	116.1	152.2
ϵ_x	in	±0.15			±0.14			±0.15		
ϵ_y	in.	±0.15			±0.14			±0.15		

Notes:

(1) $X_{c.g.}$ is Horizontal Distance Fwd (+), Aft (-) From ℄ of Vertical Thrust Engine.

(2) $Z_{c.g.}$ is Vertical Distance Above Ground Line.

3 XZ Plane is Plane of Symmetry.

4 $Y_{c.g.}$ is Lateral Distance L.H. (-). R.H. (+) From XZ Plane of Symmetry.

5 Indicates c.g. Travel if No Seat Adjustment is Made.

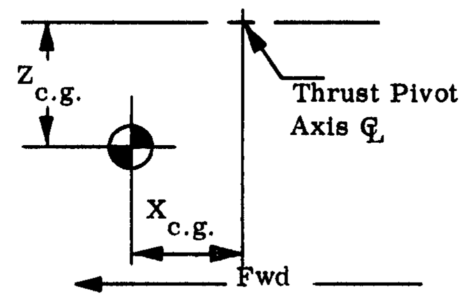


TABLE XXVIII

WEIGHT STATEMENT

Items	Weight (Lb)	Items	Weight (Lb)
Structure	(78.4)	Attitude Control	(13.4)
Tubular Framework	38.5	Thrusters - (4) at 10 lb	6.4
Honeycomb - Plume Protection	8.8	- (2) at 5 lb	3.0
Supports - Propellant Tanks	3.2	Plumbing	4.0
- Pressurization Tank	0.8	Guidance (Manual) System and Installation	8.5
- Thrust Chamber	1.4		
- Attitude Control Thrusters	4.1	Equipment (Group A Section V)	41.2
- Electronic and Communication	2.0	Furnishings	(7.0)
- Guidance	1.4	Seat Assembly (2)	6.0
- Foot	11.2	Seat Belt (2)	1.0
- Hand	2.6	Shoulder Strap	-
- Miscellaneous (Life Supt Sys)	4.4	Landing Gear Installation	36.0
Propulsion System	(65.6)		
N_2O_4 Tank Assy (2)	21.8	Crew (incl. Suit) [1] (2)	400.0
50/50 (N_2H_4-UDMH) Tank Assy (2)	21.8	Back Pack - Life Support System (2)	90.0
Plumbing and Valves	8.0		
Insulation (Tank)	14.0	Residuals - Ox., N_2O_4 (2%)	9.1
Miscellaneous	-	- Fuel, 50/50 (N_2H_4-UDMH)(2%)	5.7
Pressurization System	(20.6)	Payload	-
Tank (incl. Gas)	13.6		
Plumbing and Valves	6.0	Burnout Weight	798.7
Insulation (Tank)	1.0	Usables - Ox., N_2O_4 [2]	455.0
Engine Installation	(23.2)	- Fuel, 50/50 (N_2H_4-UDMH) [3]	285.0
Thrust Chamber (1) at 775 lb	18.0	Gross Weight	1538.7
Biprop Valve	3.1		
Heatshield	2.1		
Gimballed System	-		

[1] See Weight, Center of Gravity and Moment of Inertia Summary Sheets for Variations in the Weight of the Man.
[2] Includes 18.4 Pounds for Attitude Control.
[3] Includes 11.5 Pounds for Attitude Control.

directed toward determination of the weight penalty incurred from use of a single vehicle of the larger ΔV capability (6000 fps) off loaded for shorter missions against having optimized vehicles for smaller ΔV increments. The vehicle arrangement and mode of control was therefore maintained throughout this comparison with exception of those parameters directly affected by ΔV increases.

The latter includes tank size and arrangement and main engine thrust level to provide $T/W_i = 0.5$ which was established as a basic ground rule for transportation devices.

Table XXII, below, presents a summary of the comparison for vehicles employing a single fixed main engine arrangement and (2) 200-pound astronauts including the extravehicular suit.

TABLE XXII

VEHICLE COMPARISON SUMMARY

TWO-MAN VEHICLE - FIXED ENGINE

Design ΔV	2000	4000	6000
Gross Weight	868.8	1201.2	1538.7
Burnout Weight*	698.8	761.2	798.7

*Includes residuals = 2% of maximum load

While large weight differences exist in the gross weight of these vehicles due to propellant loading, their burnout weight difference is less than 100 pounds. In view of these results, a large two-man transportation vehicle which may be off-loaded for shorter missions seems to provide an attractive alternative for flexibility in mission objectives.

The configurations presented have been arranged to accommodate one, two, and four rigid main engines. The single engine configurations have been selected on the least weight basis and detailed weight statements and inertia tables are presented for these arrangements. In addition to the main engines, two auxiliary 50-pound thrusters are provided to permit acceleration of the vehicle in the fore-aft direction without change in the vehicle attitude. Although this arrangement is less efficient than acceleration through thrust vector positioning (changing the attitude of the vehicle to align the thrust vector or the desired component of the thrust vector with the flight path) it does maintain the vehicle attitude horizontal during the entire mission and thereby provides better visibility of the landing site, permits simplification of the sighting devices employed for navigation, and eliminates the need for gimballed or dual antennas should the necessity arise for use of radar altimeter.

The horizontal thruster arrangement is possible only in the 2000 fps ΔV vehicle shown in Figure 91 where the vertical excursions of the center of gravity are minimized by the in-line tank arrangement.

The large vertical c.g. excursions of the 4000 and 6000 fps ΔV configurations induce pitch moments of unacceptable magnitude which must be compensated by the attitude control system. Gimballing the horizontal engines to act through the instantaneous c.g. of the vehicle wastes propellants when the vertical component of the horizontal thruster in the gimballed position acts in the direction opposite to the thrust vector produced by the main lift engines.

Initial center of gravity alignment with the lift engine thrust vector, when astronauts of unequal weight board the vehicle, is provided through lateral seat adjustment. Their adjustment is made manually before lift off. A meter installed on the control panel is used to display the location of the c.g. relative to the thrust vector. The task of the astronauts before launch is to shift the seats first laterally until a zero reading is displayed on the meter provided in the operator's control panel, and then to repeat this procedure in the fore-aft direction until a zero reading is indicated on the meter.

D. ESCAPE DEVICES

The mission objective of the escape devices is to provide capability of placing one or two men into lunar orbit with minimum hardware complexity on the part of the vehicle.

Mission analyses and trajectory optimization are beyond the scope of this study; however, in order to formulate the escape device characteristics a typical ascent trajectory was assumed which is comprised of the following four phases:

(1) A boost phase - consisting of an initial continuous thrusting period.

(2) A coast phase - comprising of the period between the end of boost and initiation of the injection phase. During this period no ΔV is imparted to the vehicle; however, attitude stabilization and midcourse corrections may be required or desirable.

(3) An injection phase - consisting of one continuous or several short applications of thrust which place the escape device in position with respect to orbiting spacecraft to initiate the rendezvous maneuver, and

(4) A rendezvous phase - during which the escape device can come into close proximity with the orbiting spacecraft to enable transfer of personnel.

The ΔV requirements for such trajectories which have been established in Reference 9, range between 6000 and 7000 fps for ascent trajectories in the plane of the orbiting spacecraft. Since this condition cannot always be assumed to exist at the time of launch due to precession of the moon, and in order to provide a safety pad to accommodate deviations from the nominal flight path, a nominal ΔV capability of 8000 fps has been provided on all escape devices.

The main propulsion system requirements imposed by the escape mission are less stringent than those imposed by the transportation mission; i.e., throttling is not required, and only single mission capability is required. In addition to these, the single continuous burn time of the boost phase provides a potential application for solid propulsion systems of high specific impulse and the inherent simplicity characteristic of the solids.

Configurations of escape devices have enhanced the following variations (1) an all liquid propulsion system for the initial boost, injection, rendezvous and docking, (2) a solid propulsion system for the initial boost phase and a liquid second stage for orbital injection, rendezvous and docking, and (3) a solid propulsion system for the initial boost phase and a back pack propulsion system, modified for orbital operation to accomplish the orbit injection, rendezvous and docking.

A more detailed discussion of each system, the tradeoffs conducted within each concept and the resulting configurations are presented in the text that follows.

Figures 94 through 97 present one- and two-man escape devices with a nominal ΔV capability of 8000 fps. Although a similarity in arrangement is evident between these configurations and the previously discussed transportation devices, the following basic differences exist:

(1) The propulsion system does not provide any throttling capability.

(2) No provisions have been made for vehicle refueling since the escape vehicle has a single mission life.

(3) No landing gear has been provided.

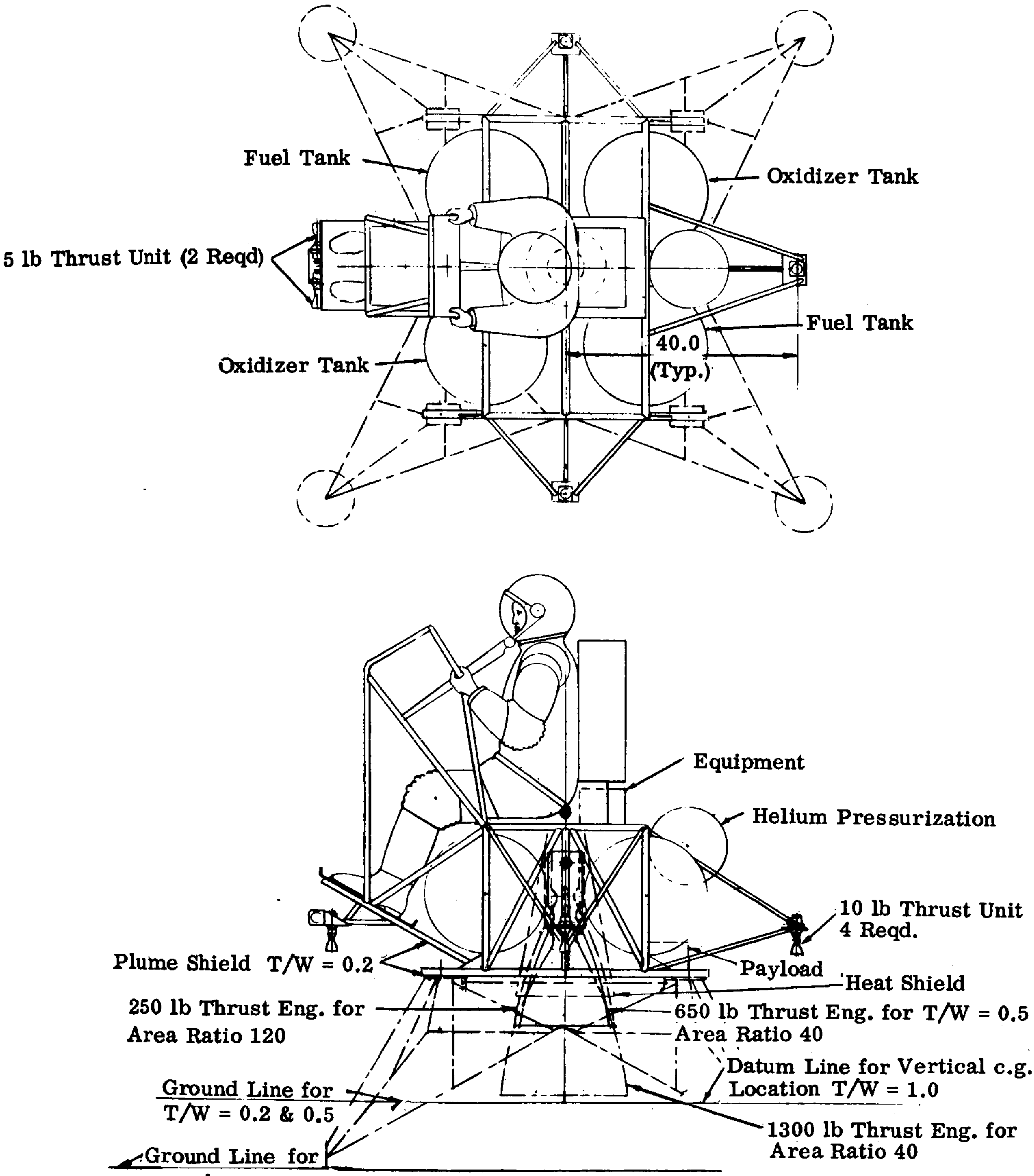
(4) The structural design criteria are based on the assumption that this vehicle would be transported to the moon in the fully fueled condition.

While throttling is requisite for precise trajectory control during the hover and landing phases of transportation missions, there is no requirement for throttling during any portion of the escape mission. Consequently only fixed thrust engines have been used on the escape device configurations.

Thrust-to-weight ratio has been shown to have a significant effect on characteristic velocity for optimized ascent trajectories (Reference 10). However, it is not obvious that the minimum complexity devices shown here would follow the same trends. Therefore, provisions have been made in the configuration studies to enable evaluation of this parameter by providing vehicles with thrust-to-weight ratios of 0.2, 0.5 and1.0, wherever possible.

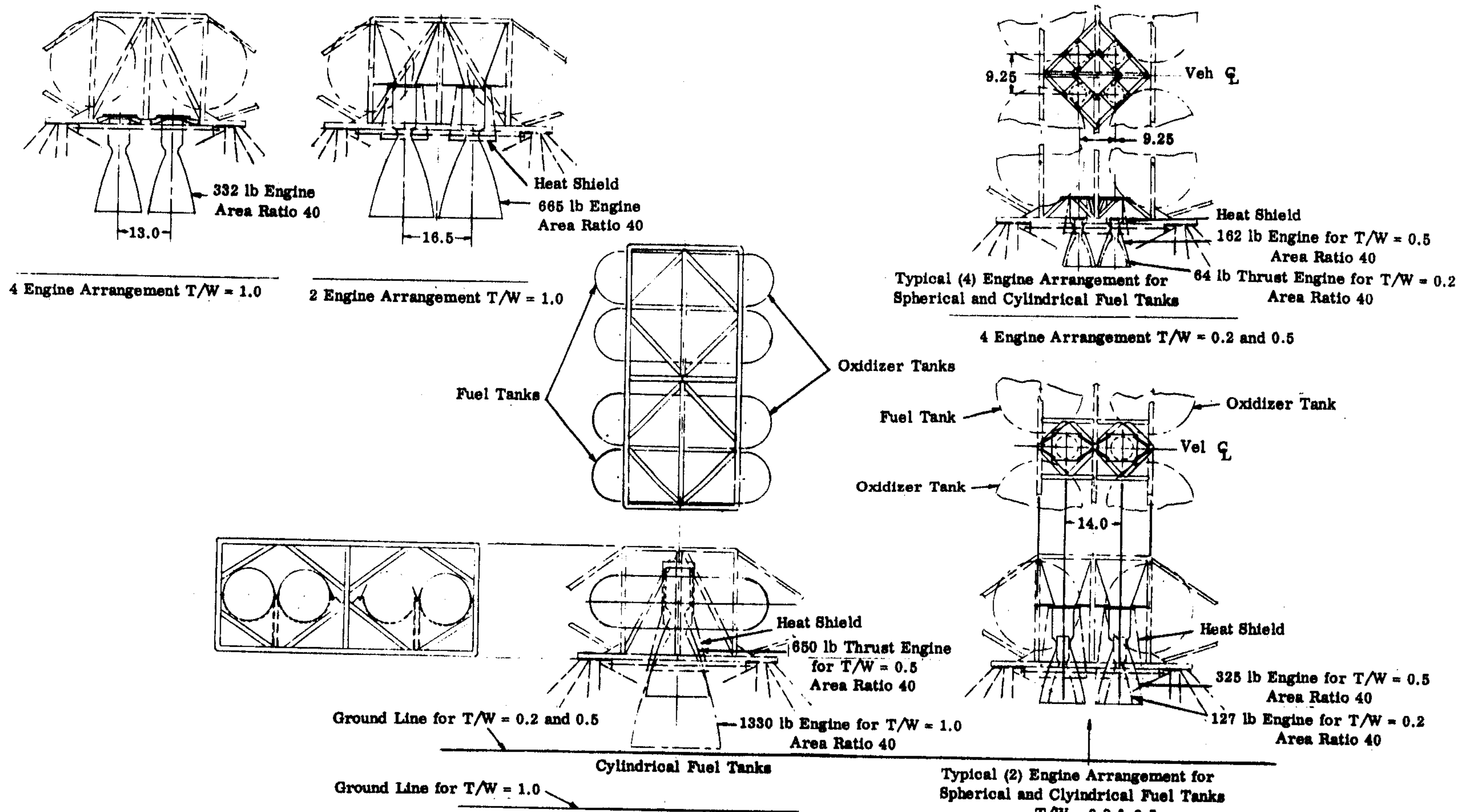
1. One Man

Figure 94 presented single-stage, one-man escape devices employing N_2O_4/50-50 propulsion system. The characteristics of these vehicles are shown in Table XXIX. Table XXXa, b and c presents a summary of the weight, center of gravity and moment of inertia for the configuration shown in Figure 94 with thrust to weight ratios of 0.2, 0.5 and 1.0. Table XXXI presents a weight statement for this



For c.g. Location & Moments of Inertial see Table No. XXX

Figure 94a. 1 Man Escape Vehicle (8000 fcs ΔV)



For c.g. Loc. & Moments of Inertia, See Table XXX

Figure 94b Tank and Engine Alternate Arrangements

TABLE XXIX

ESCAPE DEVICES

Figure No.	94			95			96			97	
				Stage I		Stage II				Stage I	Stage II
Nominal Δ V	8000 fps			5000 fps		3000 fps	8000 fps			6000 fps	2000 fps
Thrust to Weight Ratio T/W_i	0.2	0.5	1.0	0.5	1.0		0.2	0.5	1.0	0.5	
Gross Weight, Moments of Inertia, CG Location	Table XXXa	Table XXXb	Table XXXc	Table XXXIIa	Table XXXIIb		Table XXXIVa	Table XXXIVb	Table XXXIVc	Table XXXVI	B A C K
Detailed Weight Statement	See Table XXXI			Table XXXIII			See Table XXXV			Table XXXVII	
Propulsion System	H_2O_4/50-50			Solid		N_2O_4-50/50	N_2O_4/50-50			Solid	
Usable Propellants (includes R/C propellant)	1060 lb_e	1080 lb_e	1100 lb_e				1060 lb_e	1080 lb_e	1100 lb_e	1020 lb_e	
Thrust Level	250 lb_f	650 lb_f	1300 lb_f	600 lb_f	1200 lb_f	300 lb_f	370 lb_f	920 lb_f	1900 lb_f	1400 lb_f	
Chamber Pressure	80 psia	80 psia	80 psia	400 psia	400 psia	80 psia	80 psia	80 psia	80 psia	400	P A C K
→ Tank Pressure	140 psia	140 psia	140 psia			140 psia	140 psia	140 psia	140 psia		
Positive Expulsion	Teflon Bladder					Teflon Bladder	Teflon Bladder				
Chamber Characteristics	Radiation Cooled			Ablative	Ablative	Radiation Cooled	Radiation Cooled			Ablative	With
Throttling Capability											A C S
I_{sp}	308.4 sec	302 sec	304.5 sec	290 sec	290 sec	299 sec	299.8 sec	303.3 sec	305 sec	290 sec	
Area Ratio (A_e/A_t) Main Engines	120	40	40	50	50	40	40	40	40	50	
Control Mode											
Acceleration in the X Direction Acceleration in the Y Direction Acceleration in the Z Direction	Thrust Vector Positioning			Thrust Vector Positioning			Thrust Vector Positioning			Thrust Vector Positioning	
Pitch Yaw Roll	Unbalanced Moment, 6 Chamber Attitude Control System			Unbalanced Moment, 6 Chamber Attitude Control System			Unbalanced Moment, 6 Chamber Attitude Control System			Unbalanced Moment, 6 Chamber Attitude Control System	
Environmental Control System	Apollo Back Pack - 4 hr Duration			Apollo Back Pack - 4 hr Duration			Apollo Back Pack - 4 hr Duration			Apollo Back Pack - 4 hr Duration	
Landing Gear											
Equipment	Group A (see Section V)			Group A (see Section V)			Group A (see Section V)			Group A (see Section V)	
Vehicle Parameters - Gimballed Engines Gimbal Angle δ Gimbal Angle Res. $\Delta\delta$ Gimbal to CG Dist. d_G	Fixed Engines			Fixed Engines			Fixed Engines			Fixed Engines	
Max CG Excursions ϵ_x	±0.23			+0.06 -0.00	+0.06 -0.00	+13.63 -0.00	±0.21			±0.00	
ϵ_y	±0.23			+0.00 -0.01	+0.00 -0.02	+0.00 -0.04	±0.53			+0.13 -0.00	
ϵ_z	9.75			6.2	2.1	0.26	10.3			5.26	
Reaction Control System											
Commanded Acceleration	$10°/sec^2$			$10°/sec^2$			$10°/sec^2$			$10°/sec^2$	
Lever Arm Pitch	40 in.			50 in.			40 in.			38 in.	
Yaw	40 in.			50 in.			55 in.			41.5 in.	
Roll	40 in.			30 in.			40 in.			68.5 in.	
Number of Thrusters	6			6			6			6	
Thruster Size	5 lb yaw; 10 lb pitch & roll			5 lb yaw; 10 lb pitch & roll			5 lb (used in pairs except yaw)			10 lb (pitch, yaw & roll)	

TABLE XXX

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA

ONE MAN ESCAPE VEHICLE - 4 SPHERICAL PROPELLANT TANKS

(a) (1) Engine Thrust Level 260 lbs (T/W = 0.2)

		170 lb Man (Including Suit)						200 lb Man (Including Suit)					
		With Payload Propellants			Without Payload Propellants			With Payload Propellants			Without Payload Propellants		
Item	Units	0%	50%	100%	0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	475.9	841.05	1206.2	445.9	811.05	1176.2	505.9	871.05	1236.2	475.9	841.05	1206.2
$X_{c.g.}$ (1)	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Z_{c.g.}$ (2)	in.	50.99	42.78	42.16	52.74	43.40	42.57	51.37	43.27	42.52	53.02	43.89	42.93
I_{xx}	Slug ft^2	38.6	68.7	81.3	34.2	66.6	79.5	40.0	71.0	83.8	35.2	68.8	81.8
I_{yy}	Slug ft^2	50.2	80.4	93.0	42.3	75.3	88.4	52.7	84.3	97.1	44.0	78.9	92.3
I_{zz}	Slug ft^2	27.2	57.6	87.5	23.6	54.3	84.5	28.8	59.4	89.6	24.4	55.9	86.4
ϵ_x	in.	±0.21			±0.24			±0.20			±0.21		
ϵ_y	in.	±0.21			±0.24			±0.20			±0.21		

(b) (1) Engine Thrust Level = 650 lbs (T/W = 0.5)

Weight	lb	487.8	862.8	1237.8	457.8	832.8	1207.8	517.8	892.8	1267.8	487.8	862.8	1237.8
$X_{c.g.}$ (1)	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Z_{c.g.}$ (2)	in.	50.52	42.59	42.04	52.19	43.20	42.44	50.91	43.09	42.40	52.51	43.69	42.80
I_{xx}	Slug ft^2	40.6	70.4	83.9	36.3	68.3	82.0	41.9	72.7	86.4	37.4	70.6	84.4
I_{yy}	Slug ft^2	52.1	82.1	95.7	44.4	77.0	91.0	54.8	86.1	100.0	46.3	80.7	95.1
I_{zz}	Slug ft^2	27.7	58.4	89.5	24.1	55.1	86.4	29.3	60.3	91.6	25.0	56.7	88.4
ϵ_x	in.	±0.22			±0.23			±0.21			±0.22		
ϵ_y	in.	±0.22			±0.23			±0.21			±0.22		

(c) (1) Engine Thrust Level @ 1300 lbs (T/W = 1.0)

Weight	lb	503.5	888.25	1273.0	473.5	858.25	1243.0	533.5	918.25	1303.0	503.5	888.25	1273.0
$X_{c.g.}$ (1)	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Z_{c.g.}$ (2)	in.	49.74	42.27	41.82	51.32	42.86	42.22	50.17	42.77	42.18	51.68	43.36	42.57
I_{xx}	Slug ft^2	44.0	73.5	87.6	39.9	71.5	85.7	45.5	76.0	90.1	41.3	73.8	88.1
I_{yy}	Slug ft^2	55.3	85.1	99.2	47.8	80.2	94.5	58.1	89.2	103.8	49.8	83.7	98.7
I_{zz}	Slug ft^2	28.2	59.7	91.6	24.6	56.3	88.4	29.9	61.7	93.6	25.7	57.9	90.4
ϵ_x	in.	±0.21			±0.22			±0.20			±0.21		
ϵ_y	in.	±0.21			±0.22			±0.20			±0.21		

Notes:

(1) $X_{c.g.}$ is Horizontal Distance Fwd (+) or Aft (-) from ℄ of Vertical Thrust Engine.

(2) $Z_{c.g.}$ is Vertical Distance Above Ground Line

3 XZ Plane is Plane of Symmetry

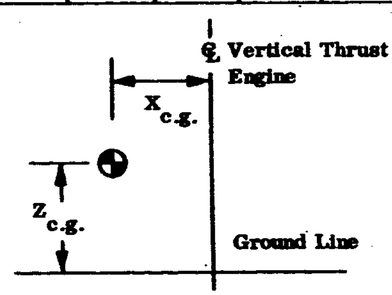


TABLE XXXI

WEIGHT STATEMENT

Items	Weight (Lb) T/W=0.2	T/W=0.5	T/W=1.0
Structure	(58.9)		
Tubular Framework	30.1		
Honeycomb - Plume Protection	11.0		
Supports - Propellant Tanks	3.2		
- Pressurization Tank	0.8		
- Thrust Chamber	1.4		
- Attitude Control Thrusters	3.1		
- Electronic and Communication	2.0		
- Guidance	1.4		
- Foot and Control	5.9		
- Hand	-		
- Miscellaneous	-		
Propulsion System	(59.6)	(62.6)	(65.5)
N_2O_4 Tank Assy	20.8	22.0	23.2
50/50 (N_2H_4-UDMH) Tank Assy	20.8	22.0	23.2
Plumbing and Valves	8.0		
Insulation (Tank)	10.0	10.6	11.1
Miscellaneous	-		
Pressurization System	(18.7)		
Tank (incl. Gas)	13.7		
Plumbing and Valves	4.0		
Insulation (Tank)	1.0		
Engine Installation	(12.0)	(20.6)	(32.8)
Thrust Chamber(s)	9.5	16.8	27.0
Biprop Valve	1.5	2.8	4.8
Heatshield	1.0		
Gimballed System	-		

Items	Weight (Lb) T/W=0.2	T/W=0.5	T/W=1.0
Attitude Control	(13.4)		
Thrusters - (4) 10 lb	6.4		
- (2) 5 lb	3.0		
Plumbing	4.0		
Guidance (Manual) System and Installation	8.5		
Equipment (Group A Section V)	41.2		
Furnishings	(4.0)		
Seat Assembly	3.0		
Seat Belt	0.5		
Shoulder Strap	0.5		
Landing Gear Installation	-		
Crew (incl Suit) [1]	200.0		
Back Pack - Life Support System	45.0		
Residuals - Ox., N_2O_4 (2%)	9.0	9.2	9.5
- Fuel, 50/50 (N_2H_4-UDMH)(2%)	5.6	5.7	6.0
Payload	30.0		
Burnout Weight	505.9	517.8	533.5
Usables - Ox., N_2O_4 [2]	449.2	462.0	474.0
- Fuel, 50/50 (N_2H_4-UDMH) [3]	281.1	288.0	295.5
Gross Weight	1236.2	1267.8	1303.0

[1] See Weight, Center of Gravity and Moment of Inertia Summary Sheets for Variations in the Weight of the Man.
[2] Includes 18.4 Pounds for Attitude Control.
[3] Includes 11.5 Pounds for Attitude Control.

configuration. The basic differences in the configurations result only from the engine size required to produce the increased thrust. The change in the overall vehicle characteristics necessary to accommodate the additional propellant for the same ΔV capability were shown to be insignificant; however, the resulting change in vehicle weight has been shown in the weight summary tables. The three configurations for thrust-to-weight ratios of 0.2, 0.5 and 1.0 are shown on the same drawing (Figure 94) with only differences in main engine shown for area ratios permitted by envelope restrictions.

The thrust level of the attitude control chambers is dictated by the thrust level of the main engines, the maximum center of gravity uncertainty anticipated in each vehicle and the control acceleration selected. Therefore the attitude thruster size depicted in each vehicle is based on the initial thrust-to-weight ratio.

Figure 95 presents a two-stage, one-man vehicle. Tables XXXII a and b present weight, c.g. and inertia summaries for thrust to weight ratios of 0.5 and 1.0. A weight statement is presented in Table XXXIII. The weight tabulation column for T/W = 1.0 displays only the weights of those components which differ from those used in the vehicle with a T/W = 0.5. The first stage consists of a solid rocket motor which provides 5000 fps ΔV. This ΔV level was selected to permit solid propellant burnout to depletion without the need for thrust termination, which in solids introduces penalties in system complexity and in addition generates an area of uncertainty in cutoff ΔV. The additional ΔV of 3000 fps required to reach escape velocity, provide midcourse corrections, orbit injection, rendezvous and attitude control for the entire mission is provided from the second stage. At depletion of the solid propellant, the motor case is ejected and the second stage of the vehicle is reoriented through a 90° pitch maneuver to align the second stage engine with the flight path.

The depicted configuration evolved from a series of designs which were considered in order to satisfy the requirements of stringent c.g. control in the plane normal to the thrust vector during stage I and stage II main engine thrusting periods.

The vehicle structure has been designed to accommodate both the 600-and 1200-pound thrust level which provides thrust-to-weight ratios of 0.5 and 1.0. Thrust-to-weight ratio of 0.2 was also investigated but was rejected because it requires burning time exceeding the operational capability of the solid motor.

The solid motor is attached to the tubular frame of the vehicle by four explosive bolts. A compressed leaf spring under each bolt provides the ejection force for stage separation after the bolts have been severed.

2. Two Man

(a) Single Stage - Liquid

Figure 96 presented a two man escape vehicle. A summary of the weight center of gravity and inertia is shown in Table XXXIV a, b and c for thrust to weight

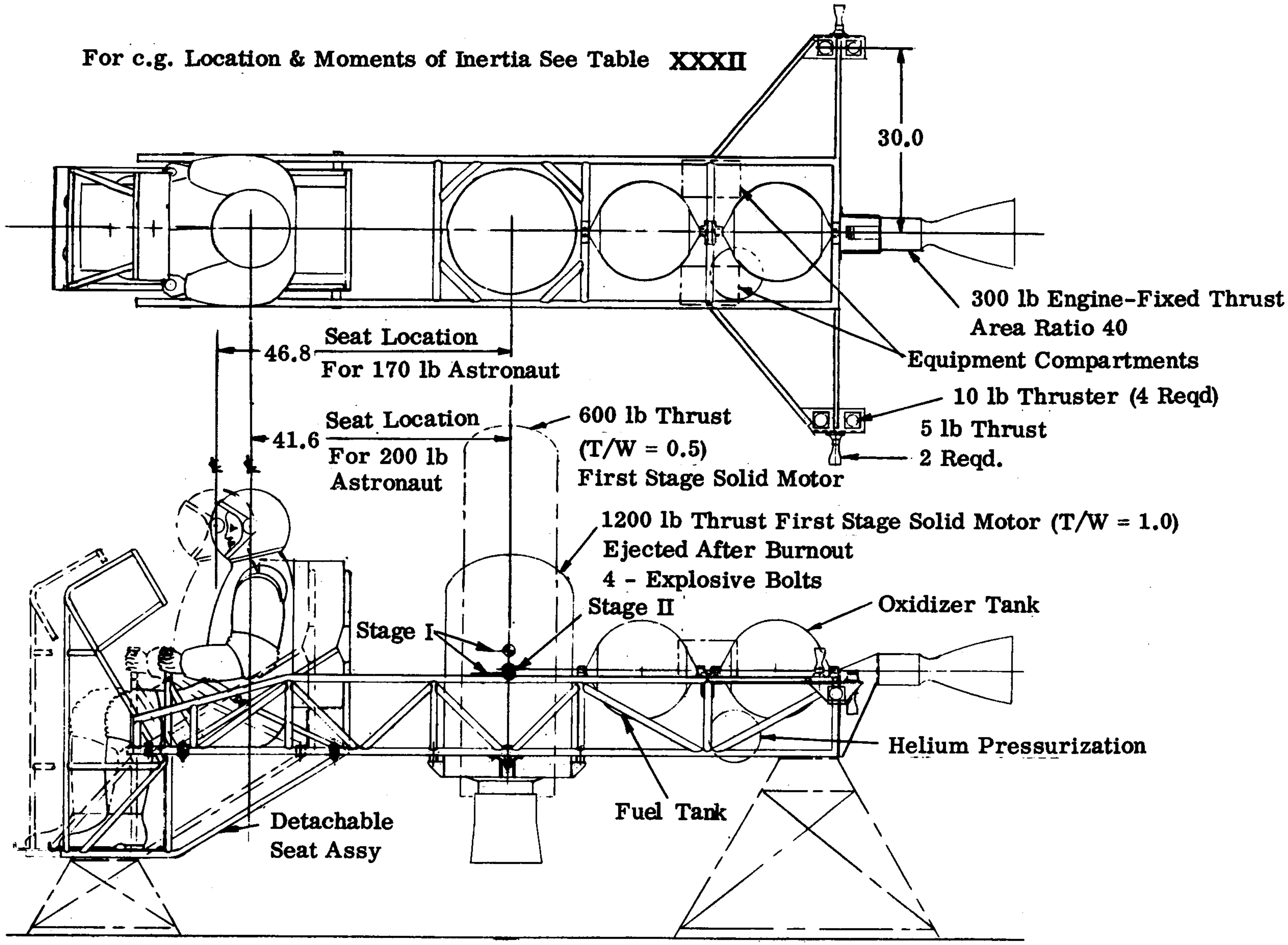


Figure 95. 1 Man Escape Vehicle Solid Propellant First Stage N_2O_4 - 50/50 Second Stage (8000 fps Δ V)

TABLE XXXII
WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA

(a) One Man Escape Vehicle - Solid Propellant Boost - T/W = 0.5

		170 lb Man (Including Suit)						200 lb Man (Including Suit)					
		Stage I & II (Boost) Propellants			Stage II (Rendezvous) Propellants			Stage I & II (Boost) Propellants			Stage II (Rendezvous) Propellants		
Items	Units	0%	50%	100%	0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	686.27	928.77	1171.27	398.33	493.30	588.27	716.27	958.77	1201.27	428.33	523.30	618.27
$X_{c.g.}$ △1	in.	0.06	0.04	0.04	14.74	5.35	0.07	0.04	0.03	0.02	13.67	5.01	0.04
$Y_{c.g.}$ △4	in.	-0.02	-0.02	-0.01	-0.04	-0.03	-0.03	-0.02	-0.02	-0.01	-0.03	-0.03	-0.02
$Z_{c.g.}$ △2	in.	43.22	49.22	46.94	44.29	44.14	44.03	43.26	49.06	46.87	44.28	44.14	44.04
I_{xx}	Slug ft^2	20.8	46.1	56.7	12.8	12.9	13.1	22.1	47.5	58.0	14.1	14.1	14.3
I_{yy}	Slug ft^2	204.9	230.1	240.7	136.7	177.2	197.1	195.9	221.3	231.8	129.1	168.4	188.1
I_{zz}	Slug ft^2	194.8	197.3	199.9	133.0	173.4	193.3	184.8	187.4	189.9	124.4	163.6	183.4
ϵ_x	in.	+0.06						+0.04					
	in.	-0.00			△5			-0.00			△5		
ϵ_y	in.	-0.01						+0.00					
	in.	+0.00						-0.02					

(b) One Man Escape Vehicle - Solid Propellant Boost - T/W = 1.0

Items	Units	0%	50%	100%	0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	678.27	920.77	1163.27	398.33	493.30	588.27	708.27	950.77	1193.27	428.33	523.30	618.27
$X_{c.g.}$ △1	in.	0.06	0.05	0.04	14.74	5.35	0.07	0.04	0.03	0.02	13.67	5.01	0.04
$Y_{c.g.}$ △4	in.	-0.02	-0.02	-0.01	-0.04	-0.03	-0.03	-0.02	-0.02	-0.01	-0.03	-0.03	-0.02
$Z_{c.g.}$ △2	in.	42.93	45.01	43.17	44.29	44.13	44.03	42.98	44.98	43.19	44.28	44.14	44.04
I_{xx}	Slug ft^2	17.9	23.8	30.8	12.8	12.9	13.1	19.2	25.0	32.0	14.1	14.1	14.3
I_{yy}	Slug ft^2	202.0	207.8	214.8	136.7	177.2	197.1	193.0	198.8	205.8	129.1	168.4	188.1
I_{zz}	Slug ft^2	195.4	200.4	205.4	133.0	173.4	193.3	185.5	190.4	195.4	124.4	163.6	183.4

Notes:

△1 $X_{c.g.}$ is Horizontal Distance Fwd (+), Aft (-) From ℄ of Vertical Thrust

△2 $Z_{c.g.}$ is Vertical Distance Above Ground Line.

3 X Z Plane is Plane of Symmetry

△4 $Y_{c.g.}$ is Lateral Distance L.H. (-), R.H. (+) from X Z Plane of Symmetry

△5 Negligible Compared to c.g. Travel Due to Propellant Consumption

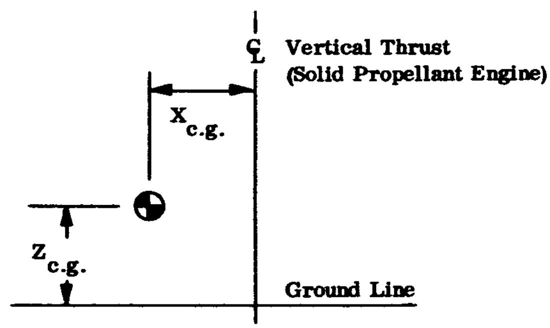


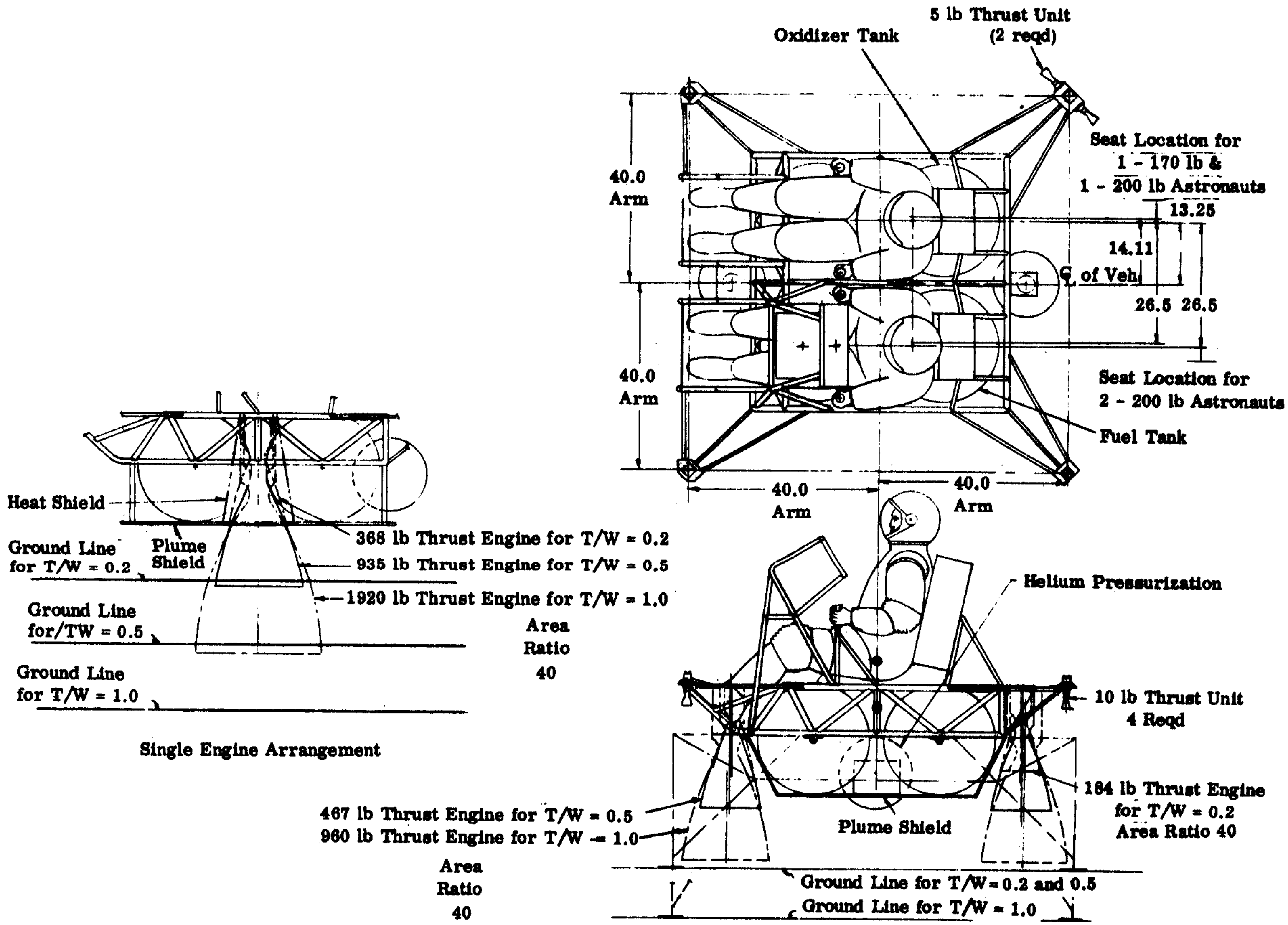
TABLE XXXIII

WEIGHT STATEMENT

Items	Weight (Lb) T/W = 0.5	Weight (Lb) T/W = 1.0
Structure	(58.20)	
Tubular Framework; Honeycomb - Plume Protection	49.30	
Supports - Propellant Tanks	1.00	
- Pressurization Tank	0.60	
- Thrust Chamber	1.50	
- Attitude Control Thrusters	2.40	
- Electronic and Communication	2.00	
- Guidance	1.40	
- Foot) Included in		
- Hand) Seat Weight		
- Miscellaneous)		
Propulsion System	(27.70)	
N_2O_4 Tank Assy	7.80	
50/50 (N_2H_4-UDMH) Tank Assy	7.80	
Plumbing and Valves	8.50	
Insulation (Tank)	3.60	
Pressurization System	(8.22)	
Tank (incl. Gas)	3.22	
Plumbing and Valves	4.00	
Insulation (Tank)	1.00	
Engine Installation	(12.15)	
Thrust Chamber	10.50	
Biprop Valve	1.65	
Heatshield	-	
Gimballed System	-	

Items	Weight (Lb) T/W = 0.5	Weight (Lb) T/W = 1.0
Attitude Control	(13.40)	
Thrusters - (4) 10 lb	6.4	
- (2) 5 lb	3.00	
Plumbing and Valves	4.00	
Guidance (Manual) System and Installation	(8.00)	
Equipment (Group A)	(41.20)	
Furnishings	(10.60)	
Seat Assembly	10.10	
Seat Belt	0.50	
Crew (incl. Suit) [1]	(200.00)	
Back Pack - Life Support System	(45.00)	
Residuals - Ox., N_2O_4 (2%)	(2.36)	
- Fuel, 50/50 (2%)	(1.50)	
Burnout Weight - Stage II, Rendezvous	428.33	
Usables - Ox., N_2O_4 [2]	115.96	
- Fuel, 50/50 [3]	73.98	
Gross Weight - Stage II, Rendezvous	618.27	
Solid Propellant Motor Case	75.00	
Solid Propellant Motor Mounting Rings	23.00	15.00
Burnout Weight - Stage I, Boost	716.27	708.27
Solid Propellant	485.00	
Gross Weight - Stage I, Boost	1201.27	1193.27

[1] See Weight, Center of Gravity and Moment of Inertia Summary Sheets for Variations in the Weight of the Man.
[2] Includes 18.4 Pounds for Attitude Control.
[3] Includes 11.5 Pounds for Attitude Control.



For c.g. Location and Moments of Inertia, see Table XXXIV

Figure 96. 2 Man Escape Vehicle 2 Engine Arrangement T/W = 0.2, 0.5 and 1.0 (8000 fps ΔV)

TABLE XXXIV

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA

TWO MAN ESCAPE VEHICLE - (4) SPHERICAL PROPELLANT TANKS

(a) (2) Engine-Thrust Level @ 184 lbs Each (T/W = 0.2)

		(2) 170 lb Men (Incl. Suit)			(2) 200 lb Men (Incl. Suit)			(1) 170 lb & (1) 200 lb Man (Incl. Suit)		
		Propellants Remaing			Propellants Remaing			Propellants Remaining		
Items	Units	0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	726.2	1256.2	1786.2	786.2	1316.2	1846.2	756.2	1286.2	1816.2
$X_{c.g.}$ $\triangle$1	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Y_{c.g.}$ $\triangle$4	in.	+0.21	+0.12	+0.08	+0.19	+0.11	+0.08	-0.33	-0.19	-0.13
$Z_{c.g.}$	in.	44.32	35.02	34.08	45.03	35.92	34.70	44.69	35.45	34.39
I_{xx}	Slug ft^2	75.6	115.4	145.4	81.2	123.4	154.6	78.4	119.5	150.1
I_{yy}	Slug ft^2	72.3	112.3	142.3	76.5	118.8	150.1	74.4	115.5	146.1
I_{zz}	Slug ft^2	64.5	106.7	150.2	68.0	110.1	153.5	66.3	108.4	151.9
ϵ_x	in.	±0.21			±0.19			±0.20		
ϵ_y	in.	+0.42 -0.00			0.38 -0.00			-0.13 -0.53		
		(b) (2) Engine-Thrust Level @ 467 lbs Each (T/W = 0.5)								
Weight	lb	745.6	1285.6	1825.6	805.6	1345.6	1885.6	775.6	1315.6	1855.6
$X_{c.g.}$ $\triangle$1	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Y_{c.g.}$ $\triangle$4	in.	+0.20	+0.11	+0.08	+0.19	+0.11	+0.08	-0.32	-0.19	-0.13
$Z_{c.g.}$ 2	in.	43.97	34.93	34.02	44.69	35.76	34.64	44.34	35.35	34.33
I_{xx}	Slug ft^2	77.2	117.3	147.4	82.9	125.3	156.8	80.0	121.4	152.1
I_{yy}	Slug ft^2	77.1	117.3	147.4	81.4	123.9	155.3	79.3	120.6	151.4
I_{zz}	Slug ft^2	68.2	111.1	155.0	71.7	114.6	158.5	69.9	112.8	156.7
ϵ_x	in.	±0.20			±0.19			±0.20		
ϵ_y	in. in.	+0.40 -0.00			+0.38 -0.00			-0.12 -0.52		
		(c) (2) Engine-Thrust Level @ 960 lbs Each (T/W = 1.0)								
Weight	lb	766.4	1316.4	1866.4	826.4	1376.4	1926.40	796.4	1346.4	1896.4
$X_{c.g.}$ $\triangle$1	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Y_{c.g.}$ $\triangle$4	in.	0.20	0.11	0.08	+0.18	+0.11	+0.08	-0.32	-0.19	-0.13
$Z_{c.g.}$ 2	in.	43.47	34.76	33.91	44.21	35.58	34.53	43.85	35.18	34.22
I_{xx}	Slug ft^2	80.1	120.2	150.7	85.9	128.4	160.1	83.0	124.3	155.4
I_{yy}	Slug ft^2	84.0	124.1	154.5	88.5	130.9	162.6	86.2	127.6	158.7
I_{zz}	Slug ft^2	72.6	116.4	161.0	76.1	119.9	164.6	74.4	118.2	162.8
ϵ_x	in.	±0.20			±0.19			±0.19		
ϵ_y	in.	+0.40			+0.37			-0.13		
	in.	-0.00			-0.01			+0.51		

Notes: $\triangle$1 $X_{c.g.}$ is Horizontal Distance Fwd (+), Aft (-) from ℄ of Vertical Thrust Engine.

2 $Z_{c.g.}$ is Vertical Distance Above Ground Line.

3. X Z Plane is Plane of Symmetry.

$\triangle$4 $Y_{c.g.}$ is Lateral Distance L.H. (-), R.H. (+) from X Z Plane of Symmetry.

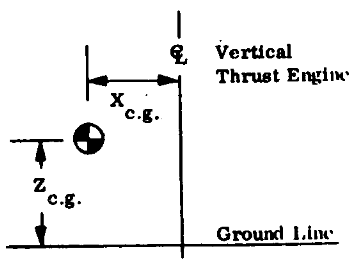


TABLE XXXV

WEIGHT STATEMENT

Items	Weight (Lb) T/W=0.2	T/W=0.5	T/W=1.0
Structure	(69.1)	(70.4)	(71.6)
Tubular Framework	26.4		
Honeycomb - Plume Protection (incl. Supports)	17.5		
Supports - Propellant Tanks	2.0		
- Pressurization Tank	0.5		
- Thrust Chamber (2)	2.4	3.7	5.0
- Attitude Control Thrusters	5.3		
- Electronic and Communication	2.0		
- Guidance	1.4		
- Foot	7.7		
- Hand	1.5		
- Miscellaneous (Life Supt Sys)	2.4		
Propulsion System	(82.4)	(84.7)	(87.0)
N_2O_4 Tank Assy	29.2	30.2	31.2
50/50 (N_2H_4-UDMH) Tank Assy	29.2	30.2	31.2
Plumbing and Valves	9.5	14.8	15.1
Insulation (Tank)	14.5		
Pressurization System	(28.0)		
Tank (incl. Gas)	22.0		
Plumbing and Valves	4.5		
Insulation (Tank)	1.5		
Engine Installation	(20.8)	(36.2)	(53.0)
Thrust Chamber(s)	15.4	29.0	42.6
Biprop Valve	2.8	4.6	7.8
Heatshield	2.6		
Gimballed System	-		

Items	Weight (Lb) T/W=0.2	T/W=0.5	T/W=1.0
Attitude Control	(13.9)		
Thrusters - (2) at 5 lb each	3.00		
- (4) at 10 lb each	6.4		
Plumbing	4.5		
Guidance (Manual) System and Installation	8.5		
Equipment (Group A)	41.2		
Furnishings	(11.0)		
Seat Assembly (2)	10.0		
Seat Belt (2)	1.0		
Shoulder Strap	-		
Landing Gear Installation	-		
Crew (incl. Suit) [1] (2)	400.0		
Back Pack - Life Support System (2)	90.0		
Residuals - Ox., N_2O_4 (2%)	13.0	13.2	13.5
- Fuel, 50/50 (N_2H_4-UDMH)(2%)	8.2	8.4	8.5
Payload	-		
Burnout Weight	786.2	805.6	826.4
Usables - Ox., N_2O_4 [2]	650.0	662.0	674.0
- Fuel 50/50 (N_2H_4-UDMH) [3]	410.0	418.0	426.0
Gross Weight	1846.2	1885.6	1926.4

[1] See Weight, Center of Gravity and Moment of Inertia Summary Sheets for Variations in the Weight of the Man.
[2] Includes 18.4 Pounds for Attitude Control.
[3] Includes 11.5 Pounds for Attitude Control.

ratios of 0.2, 0.5 and 1.0 respectively. A weight statement for these configurations is presented in Table XXXV.

Two in-line engines outside the tank cluster have been selected as the best compromise in the two-man escape vehicle using liquid propellants. The relatively large propellant loads required in this vehicle necessitate clustering of the tanks toward the center of the vehicle to reduce the effect of c.g. excursion caused by uneven drainage of the propellant tanks and to minimize the change in the moment of inertia of the vehicle from full to empty conditions, thereby minimizing the change in control power throughout the mission.

The basic configuration depicts rigid engines. Gimballing of these engines by electrical actuators can be provided at a nominal weight penalty of approximately 12% of the engine weight, plus the weight required for electrical power for actuation. However, the large engine separation distance dictates an extremely fine gimbal resolution. Consequently the rigid engine arrangement has been selected with attitude control provided by through a reaction control system.

In order to provide engine ground clearance and initial alignment, a tubular stand is provided for use as a launch platform. The platform remains on the lunar surface after launch, consequently its weight is not charged to the vehicle.

Consistent with the minimum complexity requirement, a single visual tracking cursor has been incorporated on the escape devices which the astronaut trains on a reference object. The object can be either the lunar horizon or the orbiting spacecraft with which the astronaut intends to rendezvous. Figure 97 illustrates the concept and the angular relationships associated with this reference system in the pitch plane. A cross hair type cursor arrangement would likely be used to display pitch, yaw and roll reference. When the horizon is used for pitch reference, a star near the horizon or prominent surface feature can be used as a heading or yaw reference.

During boost the horizon will drop some 20 degrees relative to the starfield, thus it may be necessary to shift the heading reference to other stars as the boost progresses. Tracking of a simple star through the boost period provides both pitch and heading reference. Roll reference would be provided by rotation of the starfield in the viewer.

Figure 98 presents a typical ascent trajectory in which the orbiting spacecraft is used as the sole reference. Again the viewer configuration can be arranged to direct the thrust at any angle relative to the line-of-sight of the orbiting spacecraft. Pitch, yaw and roll reference is provided again by a cross type cursor arrangement. In this system, the astronaut's tank is to roll about the line-of-sight to the orbiting spacecraft so that the stars are moving "up" or "down" on the viewer. Then the pitch or elevation cursor must be brought to the sighting object and held there. These are only two of several possibilities which are possible with the cursor provided.

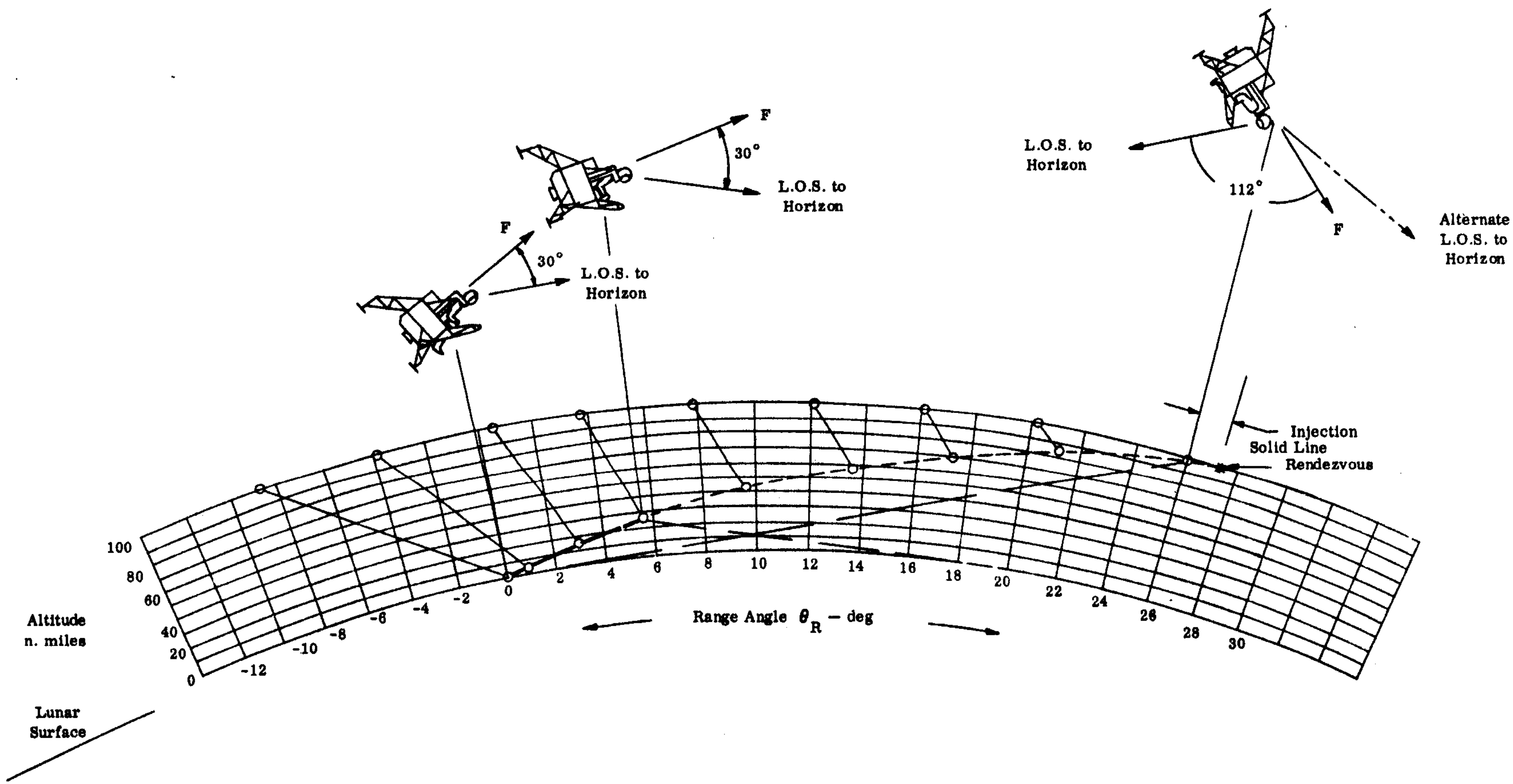


Figure 97. Lunar Escape Device Approximate Nominal Ascent Trajectory Horizon Reference

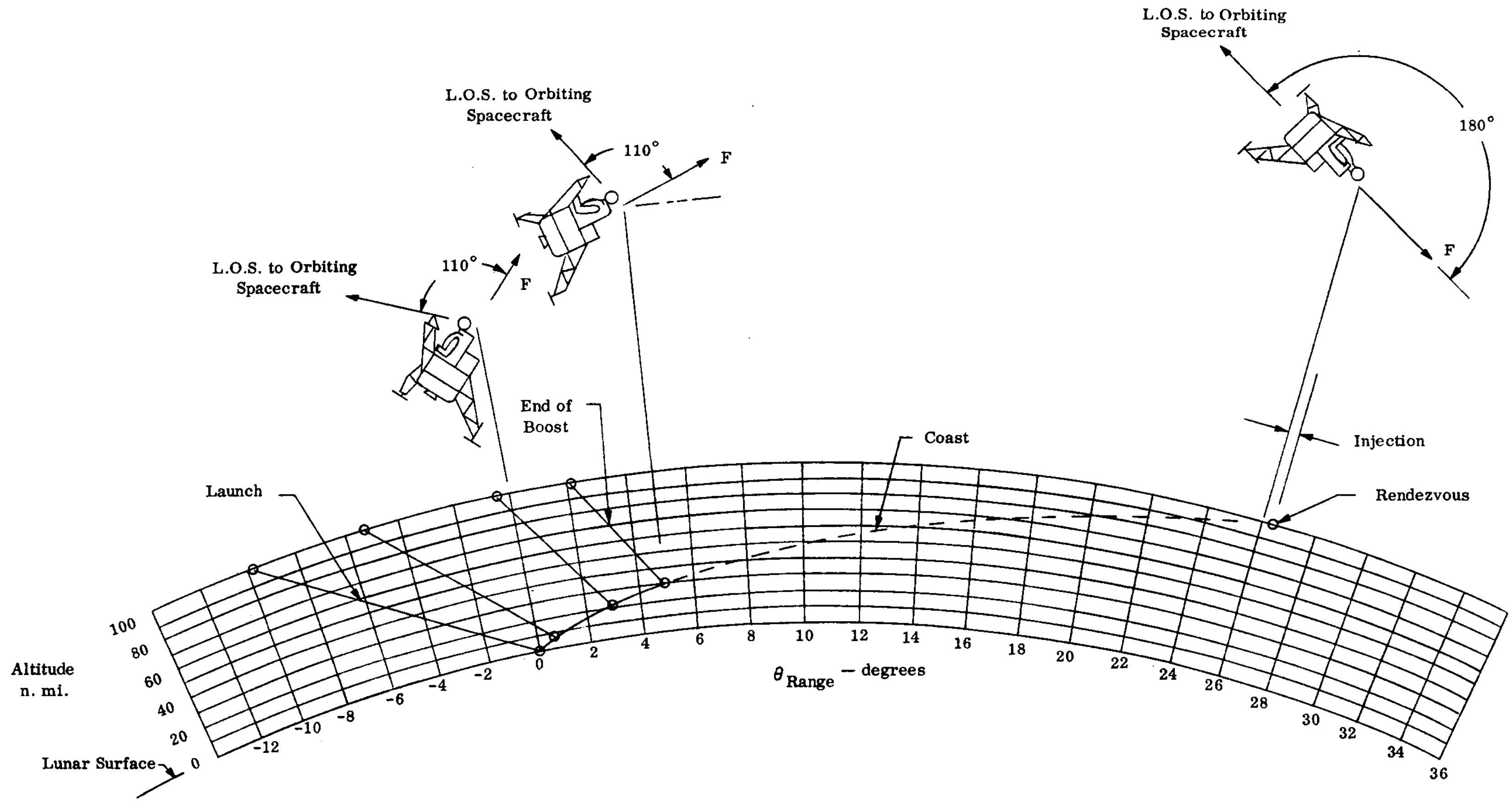


Figure 98. Lunar Escape Device Approximate Nominal Ascent Trajectory Orbiting Spacecraft Reference

In considering minimum manual systems for rendezvous and docking, it is necessary to consider the information that is needed to effect rendezvous and determine the adequacy of the information presented through direct viewing of his target. Range determination, closing velocity, closing course prediction and attitude determination represent the essential tasks which must be provided to the astronaut through visual cues. The adequacy of these,together with coupling effects of unbalanced moment producing attitude control systems and the effective thrust-to-inertia ratios presented in the variety of vehicles,can only be determined through simulation studies.

(b) Two Stage - Solid Boost, Backpack-Injection and Rendezvous

A two-stage escape device configured for two men was shown in Figure 99. The weight, center of gravity and moment of inertia for this configuration is shown in Table XXXVI; the weight statement in Table XXXVII.

The first stage consists of a solid propellant rocket motor, the tubular truss structure, and a hydrazine monopropellant reaction control system. This stage provides 6000 fps ΔV to the two astronauts each equipped with back pack propulsion system capable of delivering an additional 2000 fps ΔV. The latter comprises the second stage.

The back pack propulsion systems are essentially modifications of the one-man transportation device discussed earlier in this section. The extent of modifications to the system necessary to provide orbital maneuvering capability are itemized below.

(1) Addition of 10 attitude control chambers of and valves to provide translation and attitude control in zero "g" environment.

(2) A locking position for the gimballed thrusters so that they can be used to provide vertical thrust at minimum throttle setting.

(3) Addition of pitch-yaw-roll and translation controllers on the chest pack for orbital operation.

(4) Addition of electronic package for a rate control system with position hold feature.

The attitude stabilization and translation system for orbital operation utilizes reaction jets for angular and translation accelerations. The jets are commandable by hand controller inputs from the astronaut. The attitude system is rate commandable with an attitude hold mode similar to that shown in Figure 108(d). The pilot control commands for translation are routed through the logic to the reaction jets. These commands result in reaction jet firing to produce acceleration as long as the pilot commands persist. If these accelerations tend to induce angular rotations by producing small net moments with the c.g., the attitude control system, operating in the attitude hold mode, will pulse the reaction jets to counter the moments and maintain attitude.

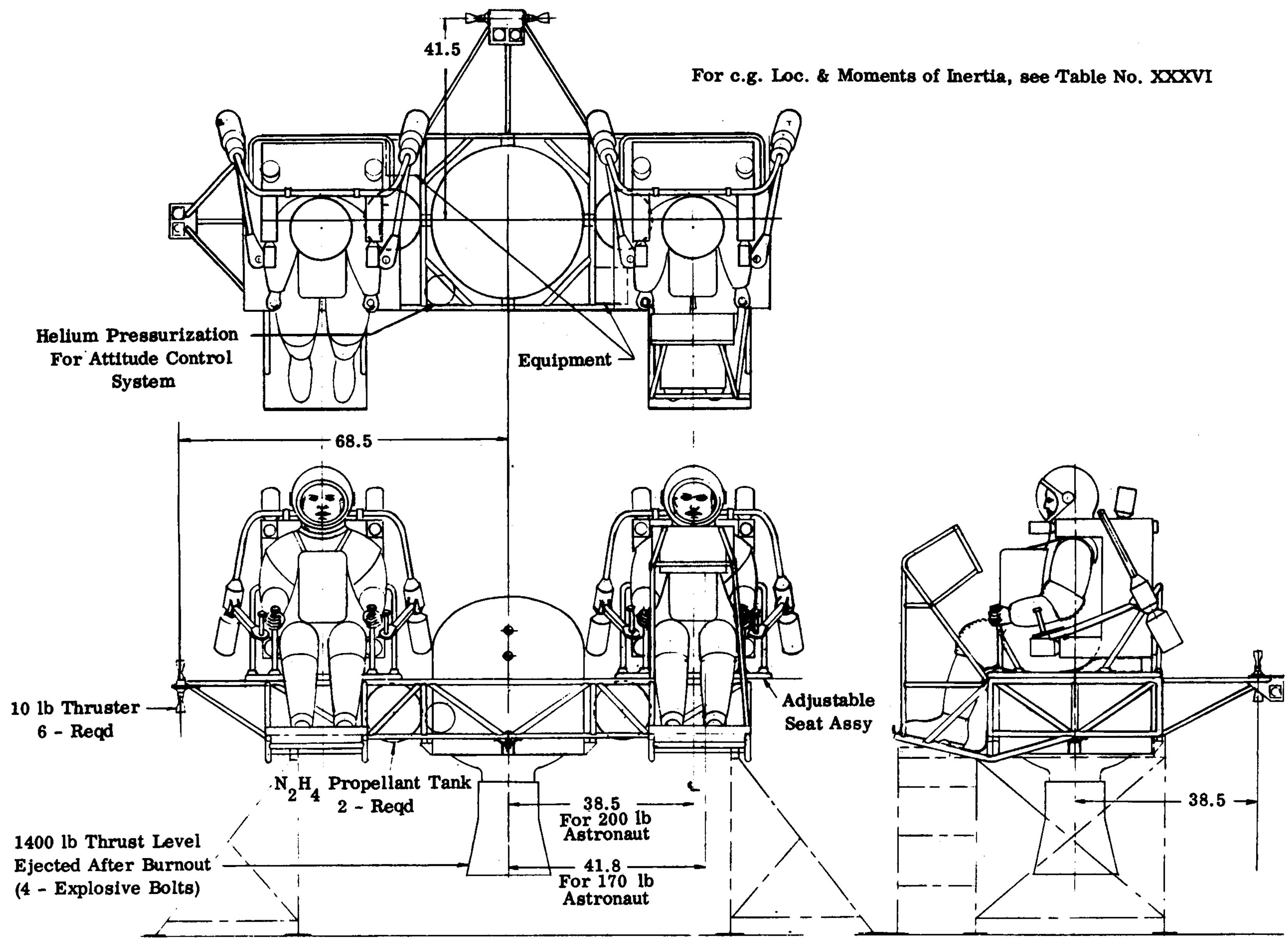


Figure 99. 2-Man (With Back Pack) Escape Vehicle Solid Propellant First Stage (8000 fps Total ΔV)

TABLE XXXVI

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA
(1) ENGINE THRUST LEVEL 1400 LB

Items	Units	(2) 170 lb Men (Incl. Suit) Propellants Remaining			(2) 200 lb Men (Incl. Suit) Propellants Remaining			(1) 170 lb & (1) 200 lb Man (Incl. Suit) Propellants Remaining		
		0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	1224.70	1762.13	2299.66	1284.80	1822.23	2359.66	1254.70	1792.13	2329.66
$X_{c.g.}$ 1	in.	0	0	0	0	0	0	0	0	0
$Y_{c.g.}$ 4	in.	0.13	0.09	0.07	0.12	0.09	0.07	0.12	0.09	0.07
$Z_{c.g.}$ 2	in.	61.51	59.49	56.25	61.70	59.69	56.49	61.61	59.59	56.37
I_{xx}	Slug Ft^2	406.9	427.7	461.3	389.0	410.1	444.3	397.9	418.9	452.8
I_{yy}	Slug Ft^2	86.4	103.8	134.0	92.2	109.9	140.7	89.3	106.9	137.4
I_{zz}	Slug Ft^2	384.0	411.7	439.4	367.2	394.9	422.6	375.6	403.3	431.0
ϵ_x	in.	±0				±0		±0		
ϵ_y	in.	+0.13 -0.00				+0.12 -0.00		+0.12 -0.00		

Notes:

1. $X_{c.g.}$ is Horizontal Distance Fwd (+), Aft (-) From ℄ of Vertical Thrust Engine. (note number printed in a triangle)
2. $Z_{c.g.}$ is Vertical Distance Above Ground Line.
3. XZ Plane is Plane of Symmetry.
4. $Y_{c.g.}$ is Lateral Distance L.H. (-), R.H. (+) From XZ Plane of Symmetry.

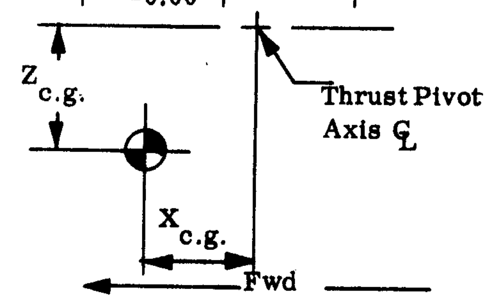


TABLE XXXVII

WEIGHT STATEMENT

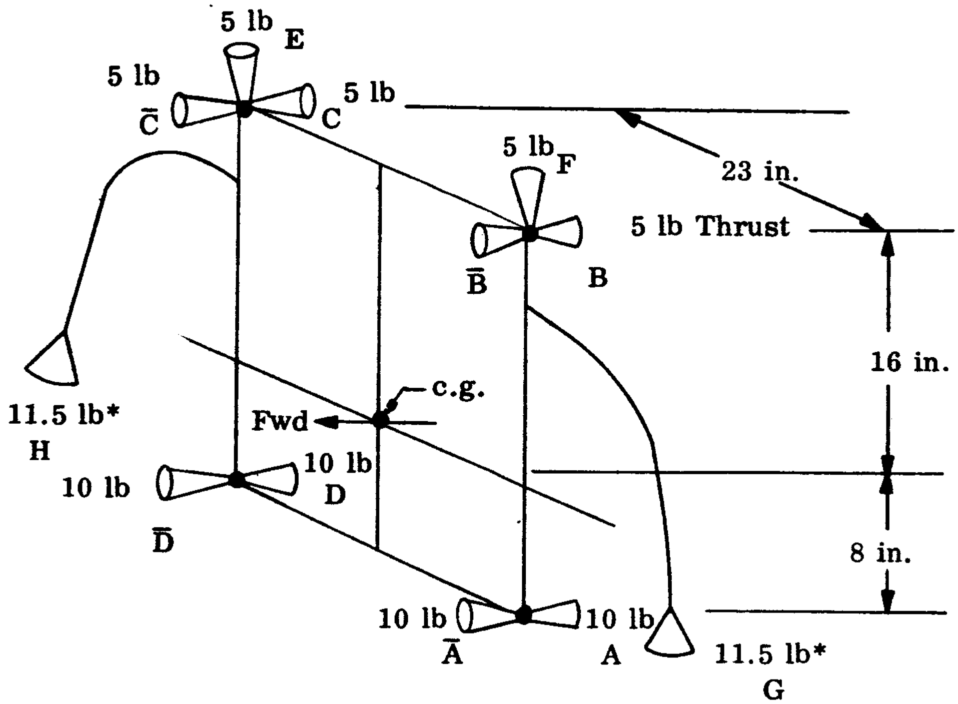
Items	Weight (lb)	Items	Weight (lb)
Structure	(101.9)	Guidance (Manual System and Installation	(8.5)
Tubular Framework	56.4		
Honeycomb - Plume Protection	8.0	Equipment (Group A Section V)	(41.2)
Supports - Propellant Tanks	1.0	Furnishings	(11.0)
- Pressurization Tank	0.3	Seat Assembly	10.0
- Solid Prop. Case Mount	17.6	Seat Belt	1.0
- Attitude Control Thrusters	2.6	Landing Gear Installation	
- Equipment	2.0		
- Guidance	1.4	Crew (incl. Suit) ⚠1 (2 at 200 lb, incl. Back	
- Foot	6.6	Pack Escape System,	
- Hand	2.0	Fwd. Pack)	(933.4)
- Miscellaneous	4.0	Back Pack - Life Support System	
Solid Rocket Installation	(160.0)	Residuals - Fuel	(1.1)
Solid Propellant Case	160.0		
		Payload	-
Attitude Control	(27.7)		
Thrusters - (6) - 10 lb	10.8	Burnout Weight	1284.8
Fuel System			
N_2H_4 Tank	4.7	Usables - N_2H_4	54.86
Insulation	1.8	- Solid Propellant	1020.00
Plumbing and Valves	2.6		
Pressurization System		Gross Weight	2359.66
Tank including Gas	3.7		
Plumbing and Valves	3.2		
Insulation (Pressurant Tank)	0.9		

⚠1 See Weight, Center of Gravity and Moment of Inertia Summary Sheets for Variations in the Weight of the Man.

An initial balance of these moments has been provided by judicious selection of the thrust level and moment arm about the nominal c.g. location.

Incorporation of a logic module to receive and implement acceleration commands and attitude correction signals in the most efficient mode is desirable. For example a pitch maneuver commanded during translation should never fire a thruster which opposes the translation direction, rather, it should shut off the appropriate thrusters for these corrections.

A schematic diagram of the attitude chambers for orbital operation is shown in Figure 100.



*11.5 lb thruster used in the transportation device set at minimum throttle.

Figure 100. Backpack Thruster Arrangement for Orbital Operations (5 df)

The additional hardware requirements in converting the surface operation backpack to one suitable for orbital operations are summarized below:

5 lb thrust attitude chamber and valves (6)	9.8 lb
10-lb thrust attitude chamber and valves (4)	6.4 lb
Additional structure and insulation	4.6 lb
Attitude and acceleration controllers (on chest pack)	1.4 lb
Attitude Stabilization system	
Gyro triad (3 axis)	2.25 lb
SCS electronics package (and logic)	5.5 lb
Power Amplifiers (4)	0.8 lb
Total Weight Increment for Modification	30.75 lb

Since this total weight of the back propulsion system modification (30.75 lb) is approximately equal to the weight of the payload capability (30.0 lb) provided for one-man operation, but not applicable in two man devices, the nominal ΔV capability is maintained.

E. DUAL FUNCTION DEVICES

Dual function vehicles are basically transportation vehicles with adequate ΔV capability to accomplish an escape mission. The transportation phase of their mission imposes the requirement for a landing gear, a refuelable propulsion system, thrust-to-weight ratio = 0.5 and throttleability requirements, while the escape phase establishes the ΔV requirements. Pertinent characteristics for dual function devices are shown in Table XXVIII.

1. One Man

Three basic arrangements were investigated for the one-man dual function devices. These include the platform arrangement shown in Figure 101a, a vehicle shown on Figure 101b, and a staged arrangement, consisting of a vehicle sized for 6000 fps, together with a back pack propulsion system modified for orbital operation capable of delivering 2000 fps ΔV, shown in Figure 101c.

The platform represents the lightest arrangement of the three systems. The savings in weight was made possible by the absence of the astronaut's seat and associated support structure which enabled a more compact vehicle arrangement.

The weight savings normally produced by vehicle staging was negated by the additional inert weight incurred from duplication of propulsion systems and equipment components. The staged vehicle gross weight including the astronaut and back pack propulsion system was estimated to be 1435 lb. Based on the results of this weight comparison, and in order to establish the applicability of kinesthetic control which is a characteristic of platform type devices, the platform arrangement shown on Figure 101a was selected for further study.

Table XXXIX presents a summary of the weight, center of gravity excursions and moments of inertia changes due to propellant consumption. Table XL presents the weight statement for the selected configuration..

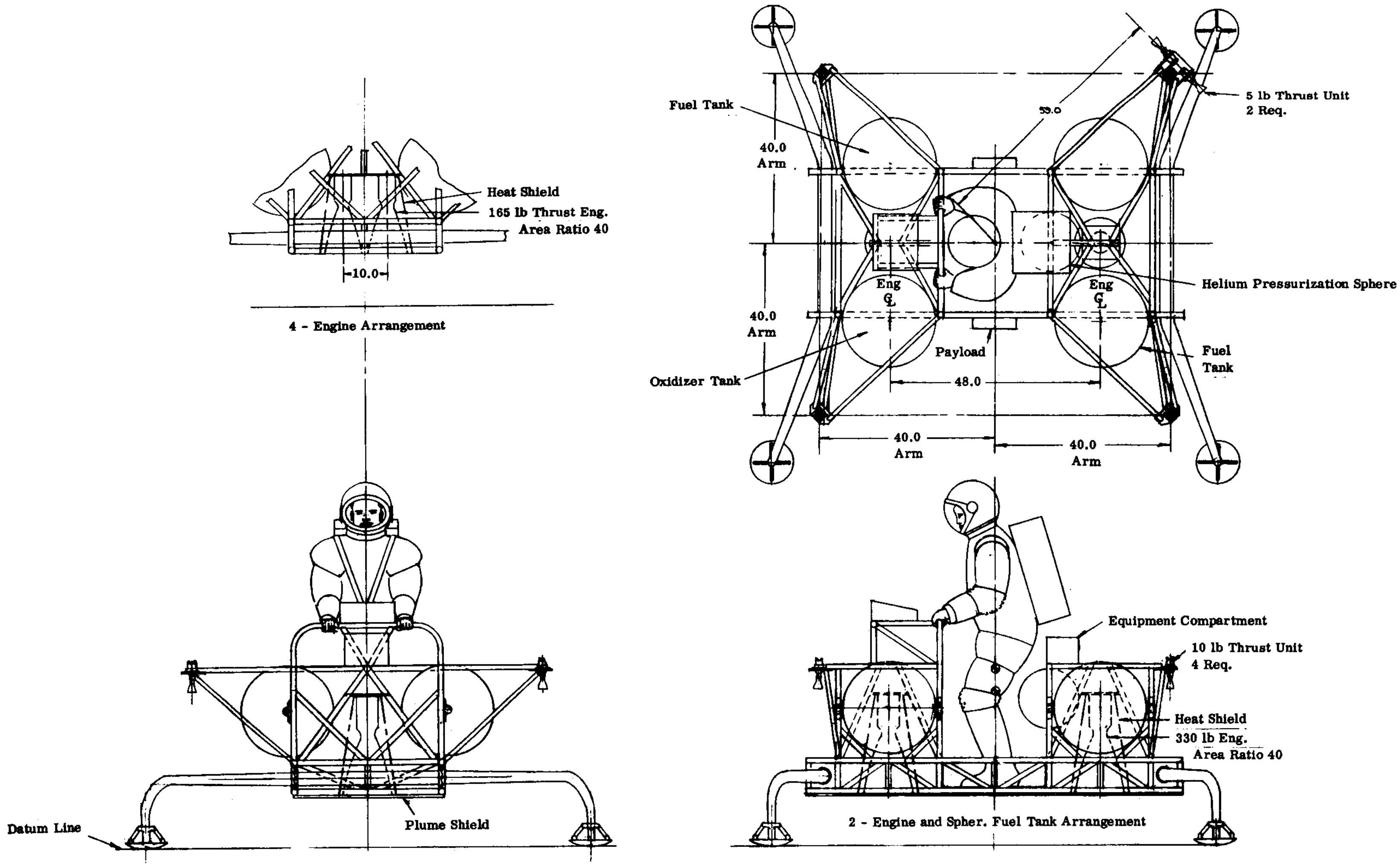
The platform employs four spherical tanks so arranged as to accommodate either two or four main engines. A single-engine configuration for the dual function platform was also considered but rejected because it elevated the location of the c.g. relative to the ground which reduces the platform stability at landing

One of the features posed for evaluation in this configuration is commanding of the vehicles attitude by means of kinesthetic control. The operator, by displacing his weight relative to the thrust vector can induce accelerations to the overall device which can be used to control the attitude of the vehicle. Since the dual function device

TABLE XXXVIII

DUAL FUNCTION DEVICES

Figure No.	101a	103a
Nominal ΔV	8000 fps	8000 fps
Gross Weight, Moments of Inertia; c.g. Location	Table XXXIX	Table XLI
Detailed Weight Statement	Table XL	Table XLII
Propulsion System		
Usable Propellants	770 lb_e	1250 lb_e
Thrust Level	660 lb_f	1000 lb_f
Chamber Pressure	80 psia	80 psia
Tank Pressure	140 psia	140 psia
Positive Expulsion	Teflon Bladders	Teflon Bladders
Chamber Characteristics	Radiation Cooled	Radiation Cooled
Throttling Capability	10:1	10:1
I_{sp}	299.4 (see insert on Fig. 50)	301
Area Ratio (A_e/A_t) Main Engines	40	40
Control Mode		
Acceleration in the X Direction	Thrust Vector Positioning	Thrust Vector Positioning
Acceleration in the Y Direction	Thrust Vector Positioning	Thrust Vector Positioning
Acceleration in the Z Direction	Throttle Control	Throttle Control
Pitch	Kinesthetic Control--see Fig. 102 or RC	
Yaw	Reaction Control	Attitude Control System 6 Chamber Unbalanced Moment
Roll	Kinesthetic Control--see Fig. 102 or RC	
Environmental Control System	Apollo ECS--4 hr duration	Apollo ECS--4 hr duration
Landing Gear	Tubular Fiber Glass 10 fps vertical equiv. landing velocity	Tubular Fiber Glass 10 fps vertical equiv. landing velocity
Equipment	Group A (see Section V)	Group A (see Section V)
Vehicle Parameters--Gimballed Engines		
Gimbal Angle δ		
Gimbal Angle Resolution $\Delta\delta$	Fixed Engines	Fixed Engines
Gimbal to c.g. Distance d_G		
Max. c.g. Excursions ϵ_x	±0.34	±0.21
ϵ_y	±0.24	±0.22
ϵ_z	5.03	
Reaction Control System		
Commanded Acceleration	$10°/sec^2$	$10°/sec^2$
Lever Arm Pitch	40 in.	40 in.
Yaw	59 in.	40 in.
Roll	40 in.	40 in.
Number of Thrusters	6	6
Thruster Size	5 lb (fixed in pairs for pitch & roll)	10 lb Fired Singly



For c.g. Loc. & Moments of Inertial See Table No. XXXIX

Figure 101a. 1-Man Escape and Transportation Platform (8000 fps Δ V)

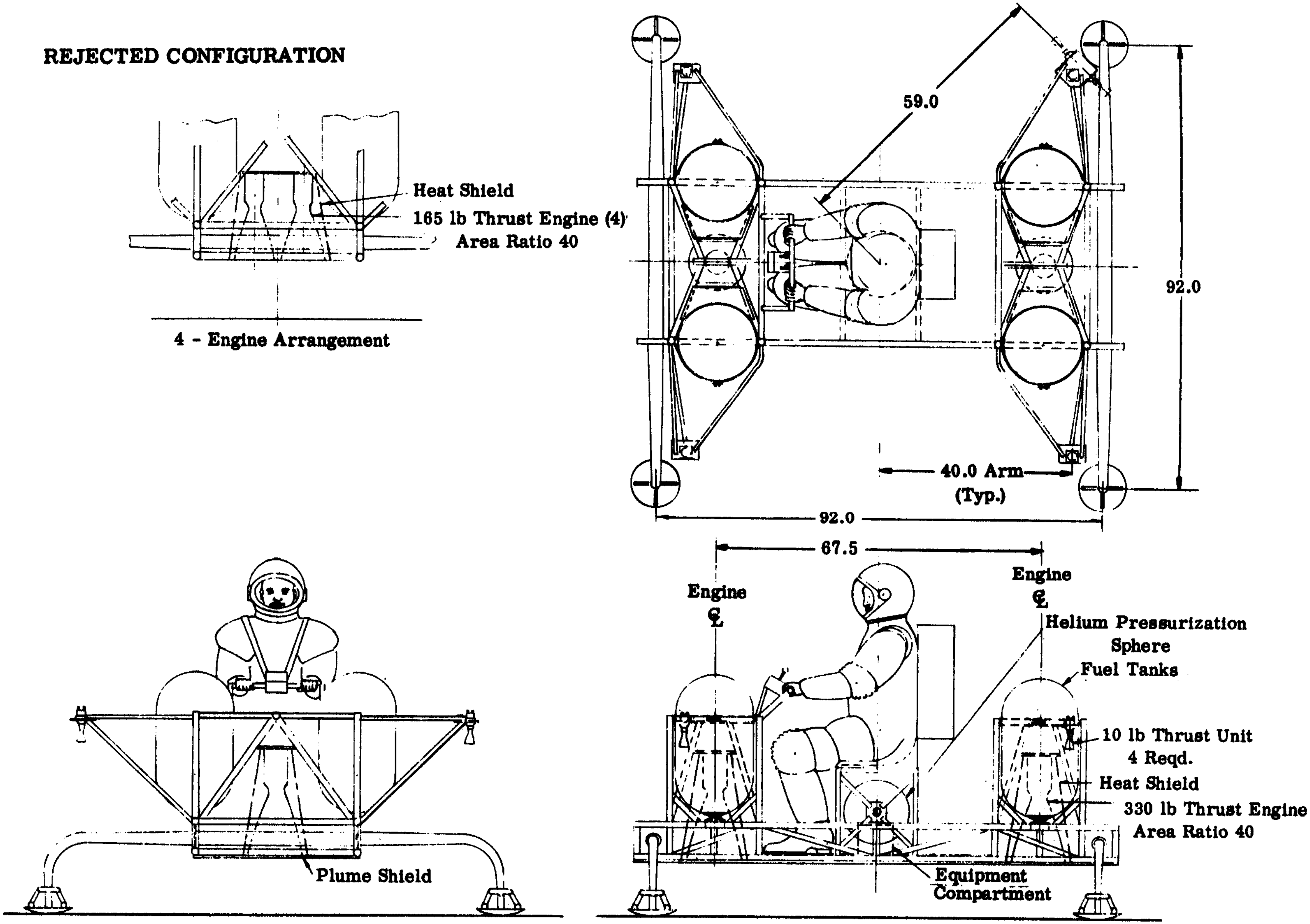


Figure 101b. Escape and Transportation Vehicle (8000 fps Δ V)

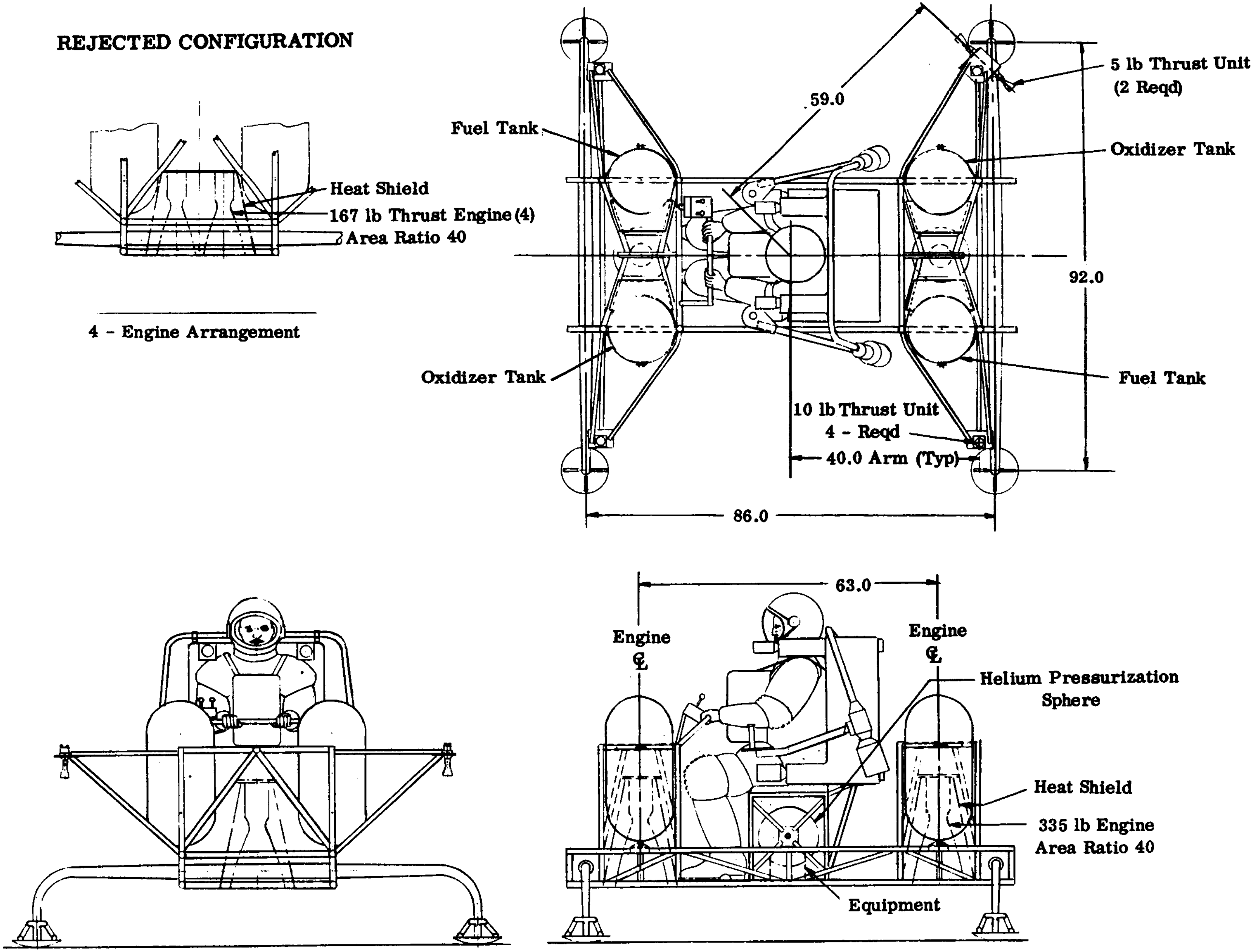


Figure 101c. Two Stage Escape and Transportation Vehicle Astronaut Equipped with Back Pack (8000 fps total Δ V)

TABLE XXXIX

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA
(1) MAN ESCAPE AND TRANSPORTATION VEHICLE - (4) SPHERICAL PROPELLANT TANKS
(2) ENGINE THRUST LEVEL AT 330 LB EACH (T/W =0.5)

Items	Units	(1) 170 lb Man (Incl. Suit) Propellants Remaining			(1) 200 lb Man (Incl. Suit) Propellants Remaining					
		0%	50%	100%	0%	50%	100%			
Weight	lb	543.8	928.8	1313.8	573.8	958.8	1343.8			
$X_{c.g.}$ 1	in.	0.00	0.00	0.00	0.00	0.00	0.00			
$Y_{c.g.}$ 4	in.	+0.32	+0.18	+0.13	+0.30	+0.18	+0.13			
$Z_{c.g.}$ 2	in.	39.89	34.85	35.17	40.60	35.43	35.57			
I_{xx}	Slug Ft^2	63.7	98.3	124.2	66.9	102.6	128.4			
I_{yy}	Slug Ft^2	87.6	133.9	171.6	90.8	138.1	175.7			
I_{zz}	Slug Ft^2	64.2	131.3	198.9	64.4	131.4	199.1			
ϵ_x	in.	±0.34			±0.32					
ϵ_y	in.	±0.24			±0.23					

Notes:

1 $X_{c.g.}$ is Horizontal Distance Fwd (+), Aft (-) From ℄ of Vertical Thrust Engine.

2 $Z_{c.g.}$ is vertical Distance Above Ground Line.

3 XZ Plane is Plane of Symmetry.

4. $Y_{c.g.}$ is Lateral Distance L.H. (-), R.H. (+) From XZ Plane of Symmetry.

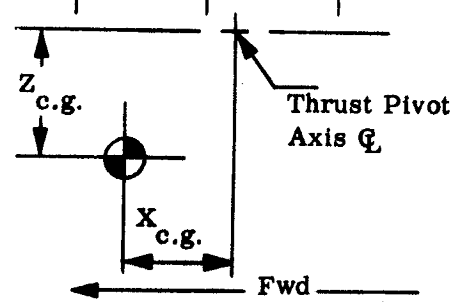


TABLE XL

WEIGHT STATEMENT

Items	Weight (Lb)	Items	Weight (Lb)
Structure	(76.5)	Attitude Control	(13.4)
Tubular Framework	48.0	Thrusters - (2) 5 lb each	3.0
Honeycomb - Plume Protection	13.3	- (4) 10 lb each	6.4
Supports - Propellant Tanks	3.2	Plumbing	4.0
- Pressurization Tank	0.8		
- Thrust Chambers (2)	2.8	Guidance (Manual) System and Installation	8.5
- Attitude Control Thrusters	5.0	Equipment (Group A)	41.2
- Electronic and Communication	2.0		
- Guidance	1.4	Furnishings	-
- Foot	-	Seat Assembly	-
- Hand	-	Seat Belt	-
- Miscellaneous	-	Shoulder Strap	-
		Landing Gear Installation	31.0
Propulsion System	(64.7)		
N_2O_4 Tank Assy (2)	22.6	Crew (incl. Suit) [1] (1) Man Standing	200.0
50/50 (N_2H_4-UDMH) Tank Assy (2)	22.6	Back Pack - Life Support System	45.0
Plumbing and Valves	8.0		
Insulation (Tank)	11.8	Residuals - Ox., N_2O_4 (2%)	9.5
Miscellaneous	-	- Fuel, 50/50 (N_2H_4-UDMH)(2%)	5.9
Pressurization System	(18.7)	Payload	30.0
Tank (incl. Gas)	13.7	Burnout Weight	573.8
Plumbing and Valves	4.0		
Insulation (Tanks)	1.0	Usables - Ox., N_2O_4 [2]	475.0
		- Fuel, 50/50 (N_2H_4/UDMH) [3]	295.0
Engine Installation	(29.4)		
Thrust Chamber (2) at 330 lb each	22.0	Gross Weight	1343.8
Biprop Valve (2)	3.6		
Heatshield	3.8		
Gimballed System	-		

[1] See Weight, Center of Gravity and Moment of Inertia Summary Sheets for Variations in the Weight of the Man.
[2] Includes 18.4 Pounds for Attitude Control.
[3] Includes 11.5 Pounds for Attitude Control.

is designed for both escape and transportation missions, this mode of control can be exploited for applicability to either or both mission profiles. The control power of this device may be determined from the instantaneous vehicle weight, commanded thrust level and the weight of the astronaut commanding the maneuver. Figure 102 presents the shift in vehicle c.g. caused by a 2-inch displacement of either a 170-lb man and a 200-pound man including his suit plus a 45-pound environmental control system.

Control for pitch and roll maneuvers may be attained without the aid of attitude thrusters during main engine firing periods. The control power at the extreme conditions, i.e., full thrust and maximum vehicle weight, and minimum thrust and burnout weight have been determined to be 43 ft-pound and 8.7 ft-pound respectively. The control power or maximum thrust seems to be adequate, or minimum thrust, which represents the hour condition near propellant depletion, the control power of 8.7 ft-pound may be marginal. The latter is equivalent to a 2.6 lb thruster or a 40 inch arm. Since yaw control cannot be implemented by kinesthetic measures, a set of two 5 pound thrusters with a nominal lever arm of 59 inches have been provided. In addition four 10 pound thrusters are installed on the vehicle so that comparison between the modes of control, kinesthetic versus reaction can be made for the platform type vehicle.

Since yaw control cannot be implemented through kinesthetic means, a set of two 5-pound thrusters with a nominal lever arm of 59 inches has been provided for yaw control. This mode of vehicle control seems feasible for trajectories requiring continuous burning of the main engines. However four additional attitude thrusters are necessary to enable vehicle stabilization during the coast period of escape trajectories or during the engine off period of ballistic flights.

Even if adequate control power is available for attitude stabilization of this platform and its implementation may be regarded as proportional rather than on-off control, the response time of the operator to displace his weight and the variation in response time with the thrust level and instantaneous inertia of the platform bring about uncertainties which again must be answered by simulation studies.

2. Two Man

Figures 103 a and b present the configurational arrangements investigated for application to the two-man devices for transportation and escape. Because the propellants represent a large fraction of the total weight in these configurations, it was necessary as in the two-man escape devices to cluster the main propellant tanks as close as possible in order to minimize the change in moment of inertia from full to empty conditions. In addition, since the dual-function devices are required to land, it is extremely important to maintain the center of gravity location as low as possible in order to affect vehicle stability at landing.

Of the configurations considered, the two-engine arrangement shown on Figure 103a was selected for further study. The single-engine arrangement could be

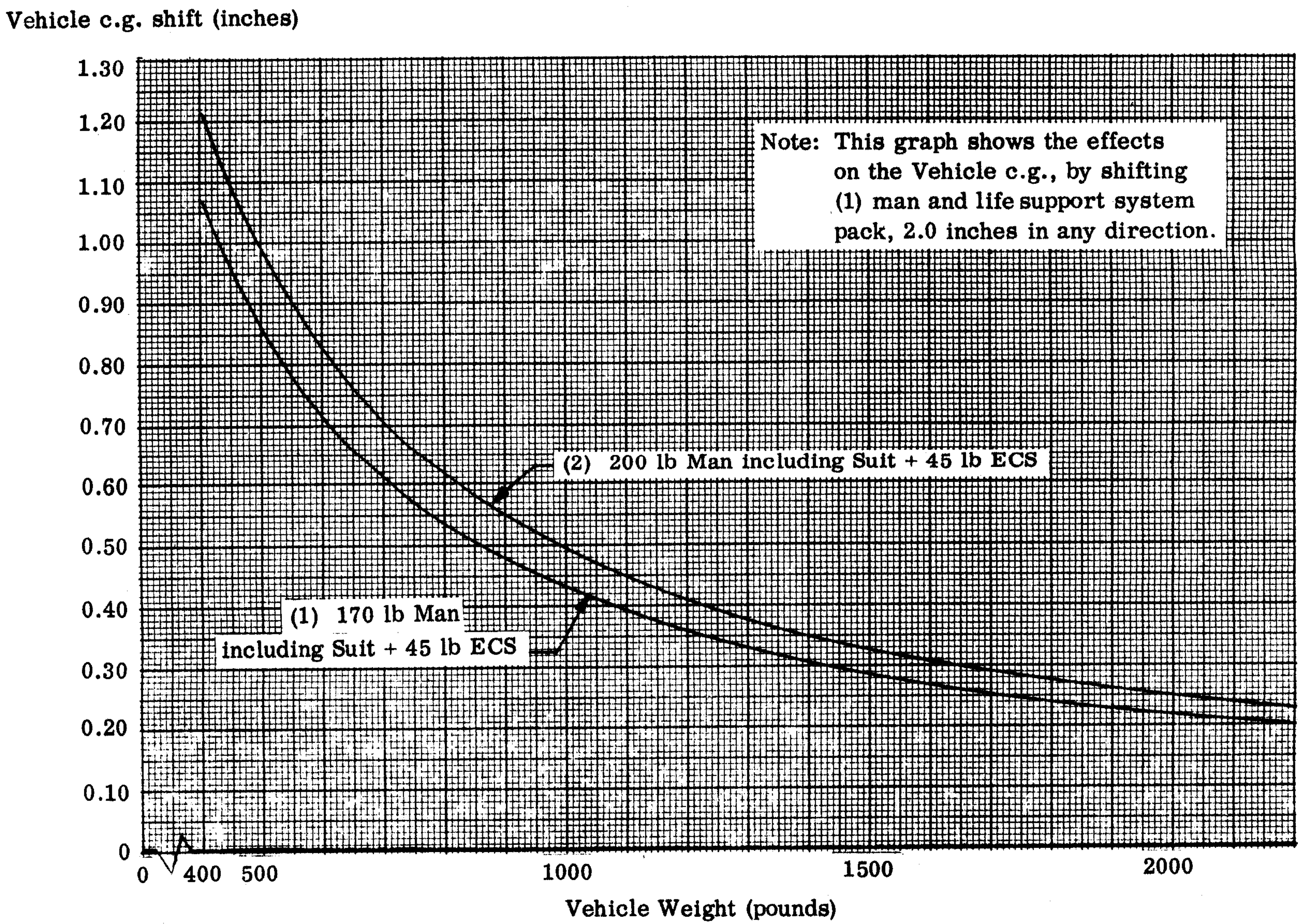
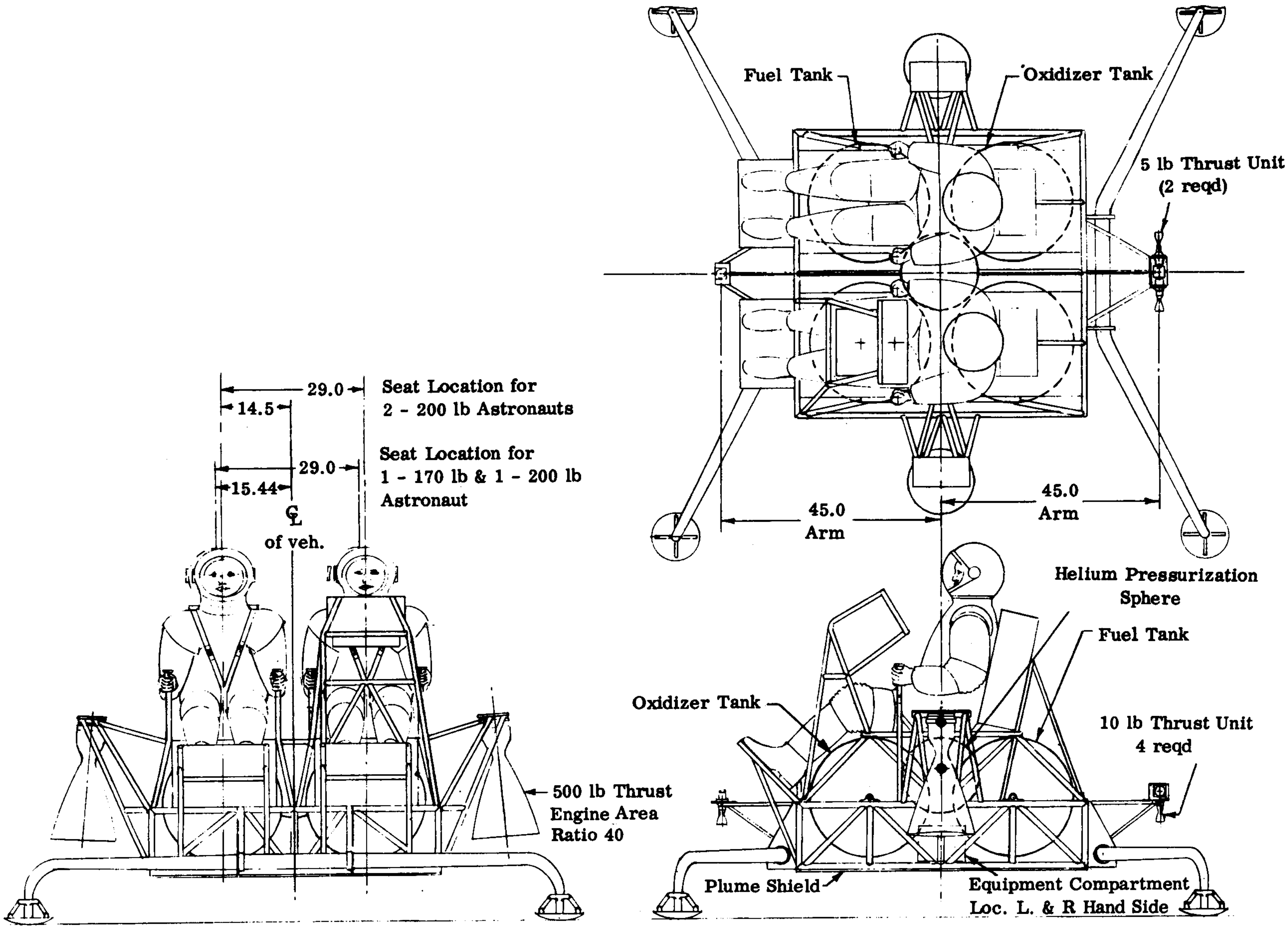


Figure 102. Vehicle c.g. Shift Due to Astronaut Weight and c.g. Variations

made competitive in vehicle dynamics only by resorting to torroidal tanks as shown in Figure 103b. However, since positive expulsion is difficult in torroidal tanks and the structural weight of the tank shell was considerably higher than that imposed by spherical tanks, the single-engine configuration was eliminated from further consideration. Table XLI presents a summary of weight, center of gravity, moments of inertia and Table XLII presents a weight statement for the configuration presented in Figure 103a.



For C6 Location & Moments of Inertia, See Table XLI

Figure 103a. 2-Man Escape and Transportation Vehicle 2 Engine Arrangement

REJECTED CONFIGURATION

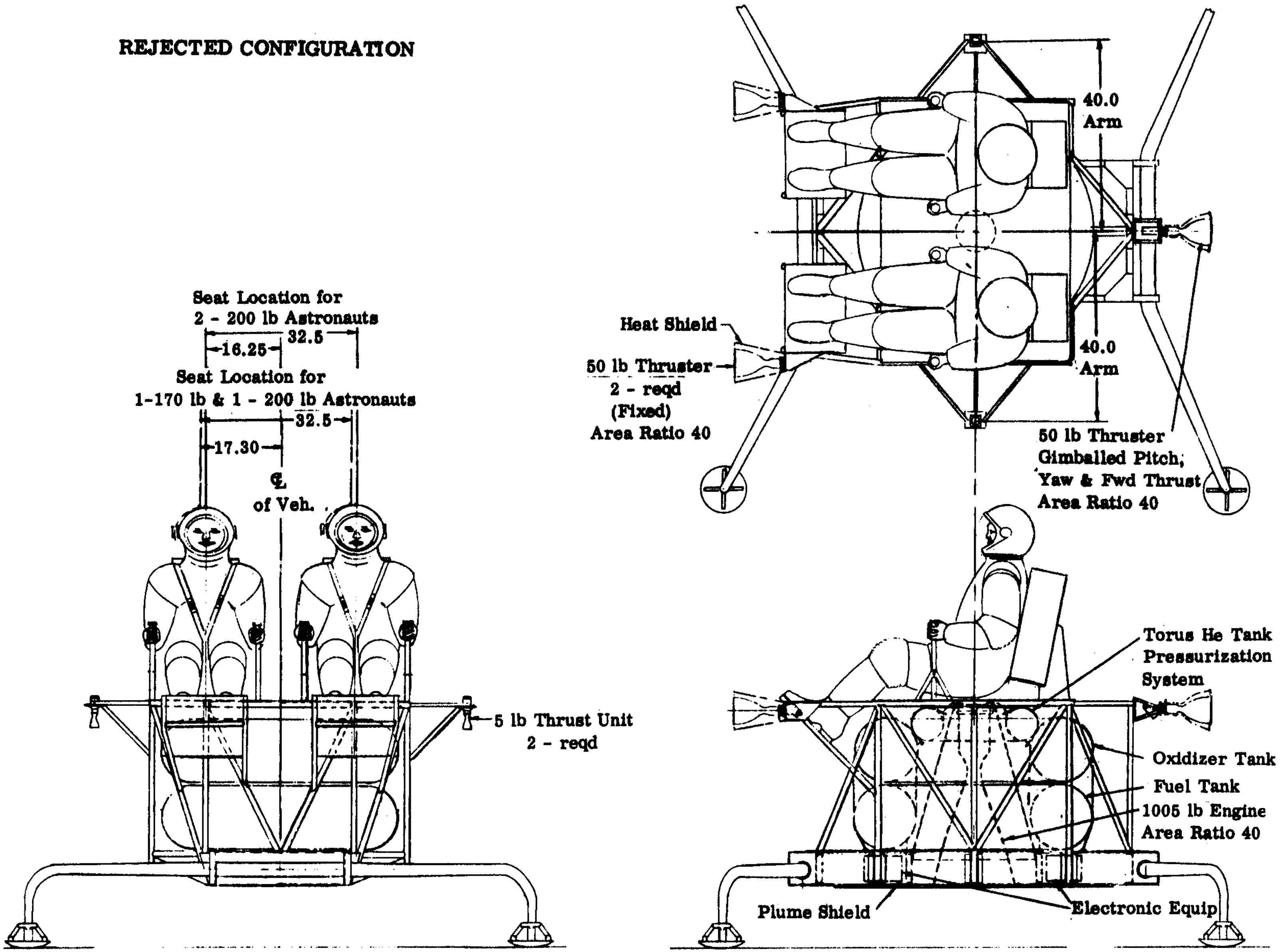
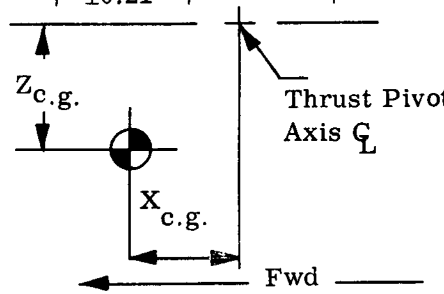


Figure 103b. Transportation and Escape Vehicle Single Engine Configuration (8000 fsp ΔV)

TABLE XLI

WEIGHT, CENTER OF GRAVITY AND MOMENT OF INERTIA
TWO MAN TRANSPORTATION VEHICLE -(4) SPHERICAL PROPELLANT TANKS
(2) ENGINE-THRUST LEVEL AT 500 LB EACH (T/W = 0.5)

Items	Units	(2) 170 lb Men (Incl. Suit)			(2) 200 lb Men (Incl. Suit)			(1) 170 lb & (1) 200 lb Man (Incl. Suit)		
		Propellants Remaining			Propellants Remaining			Propellants Remaining		
		0%	50%	100%	0%	50%	100%	0%	50%	100%
Weight	lb	826.55	1451.55	2076.55	886.55	1511.55	2136.55	856.55	1481.55	2106.55
$X_{c.g.}$ (1)	in.	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
$Y_{c.g.}$ (4)	in.	-0.01	0.00	0.00	-0.01	0.00	0.00	-0.53 [5]	-0.31 [5]	-0.21 [5]
$Z_{c.g.}$ (2)	in.	39.33	30.70	30.22	40.18	31.54	30.82	39.77	31.13	30.53
I_{xx}	Slug Ft^2	116.8	165.2	203.5	127.3	176.7	215.8	116.3	167.7	204.7
I_{yy}	Slug Ft^2	98.0	144.9	182.6	103.1	152.5	191.6	100.6	148.7	187.2
I_{zz}	Slug Ft^2	101.8	162.3	223.1	109.4	168.3	228.5	99.9	162.1	223.5
ϵ_x	in.	±0.21			±0.20			±0.21		
ϵ_y	in.	+0.20 -0.22			+0.20 -0.22			±0.21		



Notes:

(1) $X_{c.g.}$ is Horizontal Distance Fwd (+), Aft (-) From ℄ of Vertical Thrust Engine.

(2) $Z_{c.g.}$ is Vertical Distance Above Ground Line.

3 XZ Plane is Plane of Symmetry

4 $Y_{c.g.}$ is Lateral Distance L.H. (-), R.H. (+) From XZ Plane of Symmetry.

5 c.g. Shift Without Seat Adjustment

TABLE XLII

WEIGHT STATEMENT

Items	Weight (Lb)	Items	Weight (Lb)
Structure	(90.2)	Attitude Control	(13.9)
Tubular Framework	42.8	Thrusters - (4) 10 lb Thrust - incl Valve	6.4
Honeycomb - Plume Protection	14.4	- (2) 5 lb Thrust	3.0
Supports - Propellant Tanks	2.0	Plumbing	4.5
- Pressurization Tank	1.5	Guidance (Manual) System and Installation	10.5
- Thrust Chamber	9.0		
- Attitude Control Thrusters	5.9	Equipment (Group A)	41.2
- Electronic and Communication	2.0		
- Guidance	1.4	Furnishings	(11.0)
- Foot	5.3	Seat Assembly (2)	10.0
- Hand	1.5	Seat Belt (2)	1.0
- Miscellaneous	4.4	Shoulder Strap	-
		Landing Gear Installation	42.95
Propulsion System	(94.4)		
N_2O_4 Tank Assy	34.7	Crew (incl. Suit) [1] (2)	400.0
50/50 (N_2H_4-UDMH) Tank Assy	34.7		
Plumbing and Valves	9.5	Back Pack - Life Support System (2)	90.0
Insulation (Tank)	15.5	Residuals - Ox., N_2O_4 (2%)	15.4
Miscellaneous	-	- Fuel, 50/50 (N_2H_4-UDMH)(2%)	9.6
Pressurization System	(31.2)	Payload	-
Tank (incl. Gas)	25.3		
Plumbing and Valves	4.5	Burnout Weight	886.55
Insulation (Tank)	1.4	Usables - Ox., N_2O_4 [2]	770.0
		- Fuel, 50/50 (N_2H_4/UDMH) [3]	480.0
Engine Installation	(36.2)		
Thrust Chamber (2) 500 lb thrust	29.0	Gross Weight	2136.55
Biprop Valve (2)	4.6		
Heatshield	2.6		
Gimballed System	-		

[1] See Weight, Center of Gravity and Moment of Inertia Summary Tables for Variations in the Weight of the Man.
[2] Includes 18.4 Pounds for Attitude Control.
[3] Includes 11.5 Pounds for Attitude Control.

A. SUMMARY

The vehicle configurations and control characteristics presented in this report are intended to provide the data necessary to conduct guidance and control simulation studies of representative lunar transportation and escape vehicles. This section briefly discusses the general form of such a simulation, the use of the various tables and figures of this report as sources of parameter values for inclusion in the simulation, and the use of the simulation to refine the vehicle control characteristics on the basis of specific mission requirements. Simulation considerations pertaining exclusively to the back pack type of vehicle are also discussed.

B. SIMULATION BLOCK DIAGRAMS

The general form of a simulation of the manned vehicle is shown in Figure 104. Pilot inputs are typically applied to the vehicle through some type of a guidance

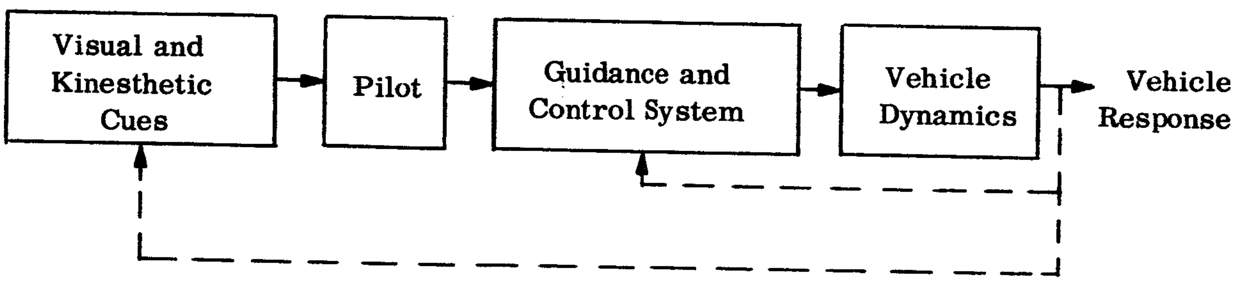


Figure 104. General Simulation Block Diagram

and control system, which may range from a very complex system utilizing radar and an inertial platform, to a simple rate gyro system. Direct manual input to the moment producing devices is also possible, although its use is frequently restricted to a backup mode of operation. The vehicle control system and inertial characteristics, in conjunction with the physical equations of motion, determine the vehicle response, which is fed back to the pilot through visual and kinesthetic cues (in the case of a moving base simulator), and is also fed into the guidance and control system portion of the simulation to close the automatic loop.

The "Vehicle Dynamics" block of Figure 104 is expanded in the block diagrams of Figures 105, 106, and 107. These diagrams represent vehicles which employ only one of the three basic means for producing pitching and rolling attitude moments:

A gimballed main propulsion unit for lift and translation as well as 2-axis attitude maneuvering. (Figure 105)

A rigidly mounted four main thruster configuration, where all four engines are independently throttleable, for lift and translation as well as 2-axis attitude maneuvering. (Figure 106)

A rigidly mounted main propulsion unit for lift and translation and a set of 8 reaction jets for 3-axis attitude maneuvering. (Figure 107)

These three configurations formed the basis for the various requirements, constraints, and relationships presented in Section II, which were developed for the screening and selection of the various vehicle configurations. Figures 105, 106, and 107 depict in a basic form the various attitude and translational control system components and subsystems and their functional relationships. Indicated on these diagrams are additions, multiplications, trigonometric operations, nonlinear effects, and inputs from the monitoring and up-dating unit of the simulation. All three systems represented by these diagrams employ vehicle-body axis coordinates, with the orthogonal coordinates originating at the vehicle center of gravity.

Figure 105 shows the diagram representative of a vehicle which uses only the gimballed main propulsion unit for pitch and roll attitude maneuvering. The yaw axis portion of this system, not shown in Figure 105, is identical to that of Figure 107. Indicated on this diagram are the various possible coupling effects which may occur when the c.g. is shifted from its nominal location. Indicated also, along the top of this diagram, are the sources where the required parameters, constants, limits, etc. can be obtained which will determine the particular vehicle configuration to be simulated

Figure 106 shows the diagram representing a vehicle which uses only four throttleable main engines for pitch and roll attitude maneuvering. The yaw axis portion of this system is again identical to that of Figure 107. As in Figure 105, this diagram shows the coupling effects due to c.g. shifts and the required characteristic parameters, constants, etc. The four summing points to the right of the control inputs illustrate the principle of collective and differential throttleability. Figure 106 is valid for a 4-engine "diamond" main engine configuration; a 4-engine "square" main engine configuration would have only a slightly different summing point arrangement. (For a discussion of multiple main engine configurations see Section II).

Figure 107 shows the diagram representative of a vehicle which uses only a set of 8 reaction jets for attitude maneuvering. Coupling effects due to c.g. shifts are indicated by the summing of various forces and moments. The required characteristics, parameters, and constants are listed along the top of the diagram.

Vehicle control systems are not shown in Figures 105, 106, and 107; these are required to close the loop (for each axis) from the output of the system to the control inputs. In the simplest case this would consist of merely displaying to the pilot some or all of the system outputs. Block diagrams of various single-axis control loops are

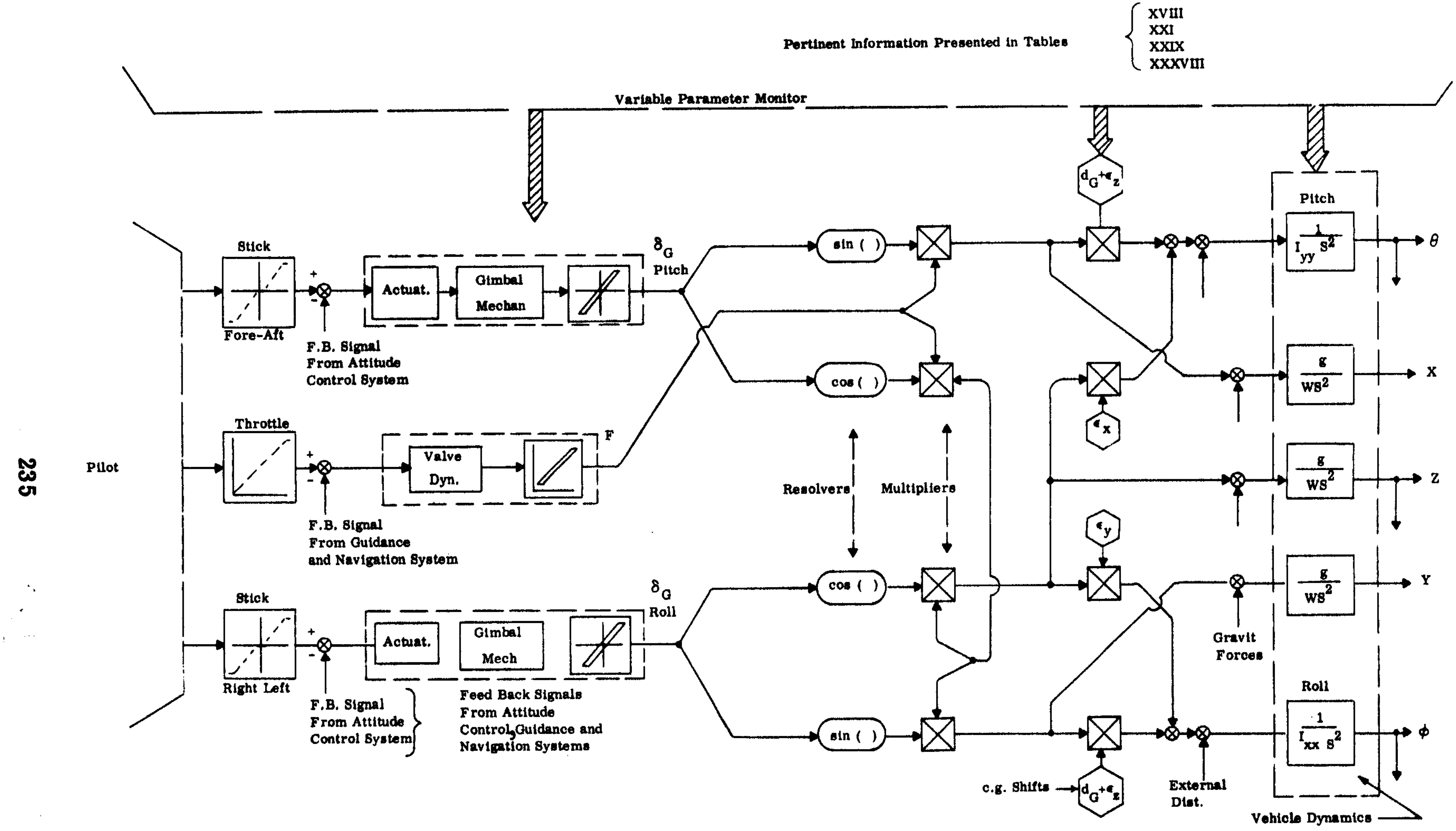


Figure 105. Configuration Using Only a Gimballed Main Propulsion Unit Used for Attitude Control

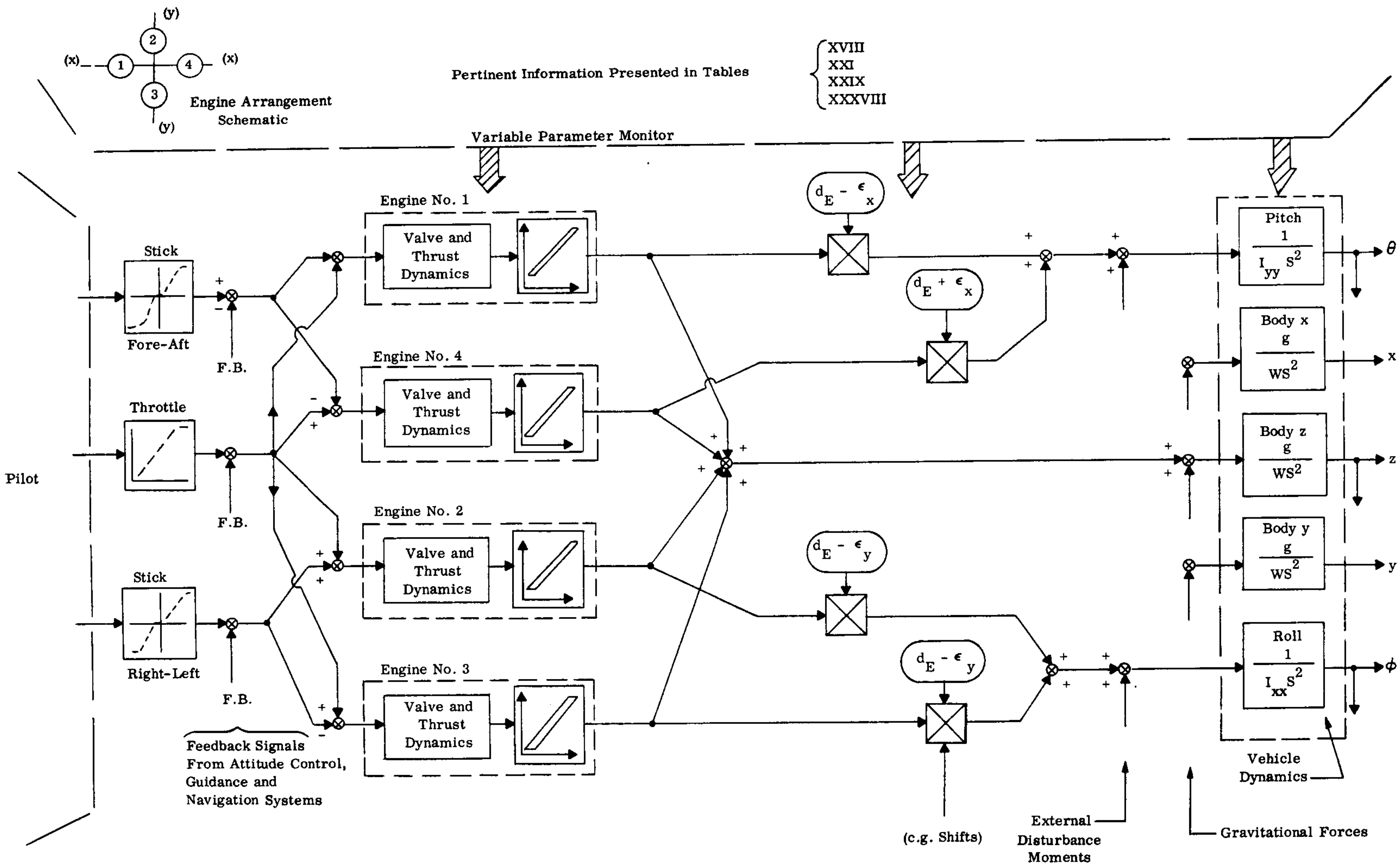


Figure 106. Four Main Thruster Configuration Using Only Differential Throttling for Attitude Control

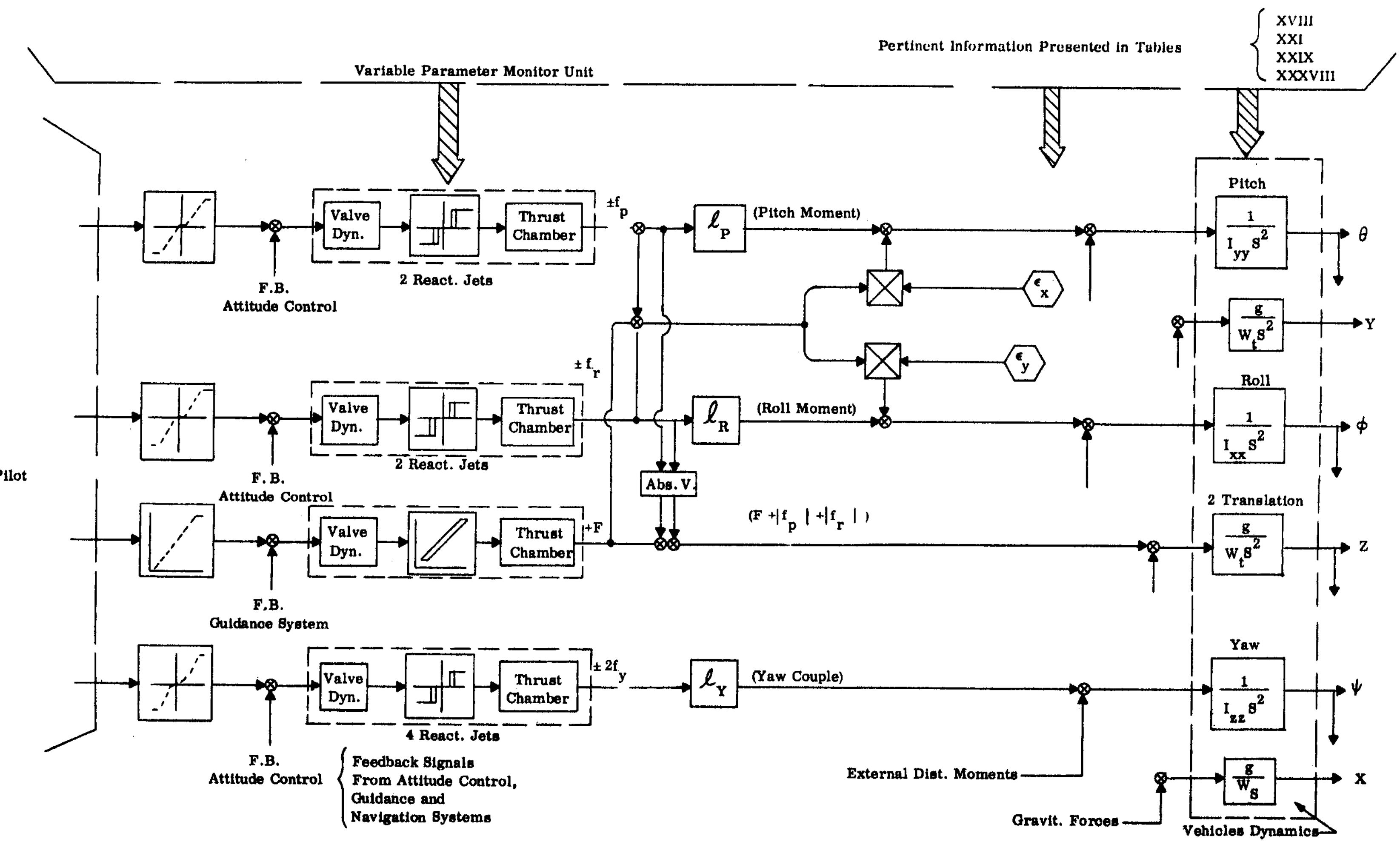
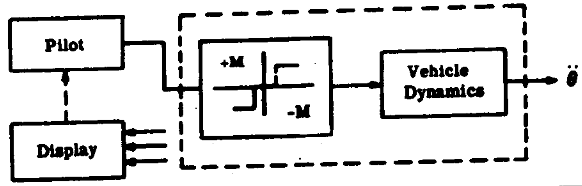
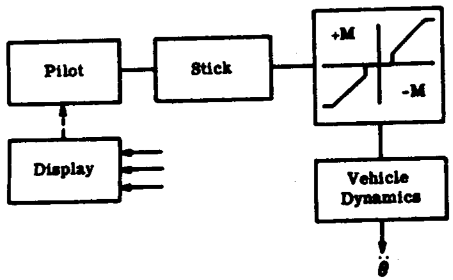
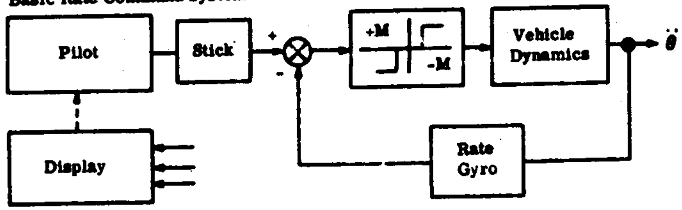


Figure 107. Configuration Using Only a Set of 8 Reaction Jets for Attitude Control

Loop		Components/Weight	Power Ave	Power Peak	Remarks
a. On-Off Acceleration  		(3) Switches @ 0.125 0.375 Pitch Yaw Lever 0.5 Roll Lever 0.5 (6) Valves @ 1.3 (7.8)* Wire & Solder 0.18 9.35	~0	168W	* On-Off Valves are Integrated with Thrust Chambers and are Included in the Vehicle Weights
b. Angular Acceleration Proportional to Stick Deflection  	(1) Throttling of R. C. Thrust	(3) Input Amplifiers 0.18 (3) Power Amplifiers 0.56 (6) Proportional Valves (9.60)* Pitch Yaw Lever 0.5 Roll Lever 0.5 (3) Pot Pickoffs 0.188 Brackets, P.C. Boards, etc. 0.5 Power Supply 1.2 13.23	~5W	-173W	* 1.8 lb Increase to Total Thruster Weight Above That Shown on Vehicle Configurations
	(2) Gimballing of R. C. Thrust	(3) Input Amplifiers 0.18 (3) Solenoid Valves 3.9 (3) Power Amplifiers 0.56 Pitch Yaw Lever 0.5 Roll Lever 0.5 (6) Pot Pickoffs 0.375 Brackets, Dividers, P.C. Boards 0.5 GimbalLinkages & Actuart. 6.0 Power Supply 0.2 12.71	~180W	-180W	Impractical - Requires 180° Gimballing of the Thrust Chamber, Flexible Lines and Actuators
	(3) PWM of Above System	(3) Input Amplifiers 0.18 (6) Solenoid Valves 7.8 (3) PWM Networks 0.31 (3) Power Amplifiers 0.56 Pitch Yaw Stick 0.5 Roll Lever 0.5 (3) Pot Pickoffs 0.188 Brackets, Dividers, P. C. Boards, etc. 0.5 Power Supply 0.2 10.74	~5W	173W	
c. Basic Rate Command System  		Same as b(3) 10.74 (3) Rate Gyros 2.25 12.99	~9W	177W	

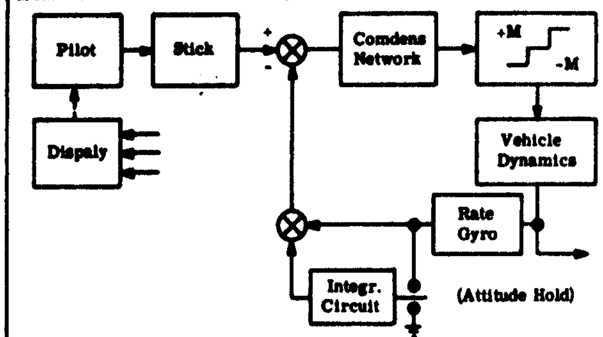
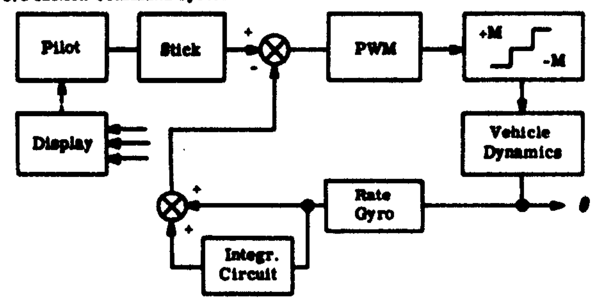
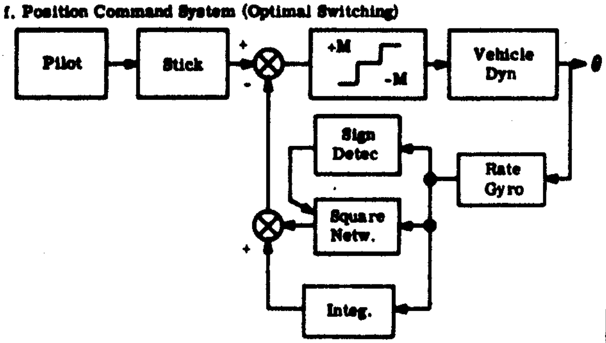
Position Hold	Components/Weight	Power Ave	Power Peak	Remarks
d. Rate Command Position Hold System 	Weight of (c) 12.99 (3) Integrators 0.1 (3) Compensation Netwk 0.18 13.27	9 W	177 W	
e. Position Command System 	Same as d	9 W	177 W	
f. Position Command System (Optimal Switching) 	Weight of b 12.99 (3) Squaring Circ. 0.18 13.17	9 W	177 W	

Figure 108. Block Diagrams for Various Single Axis Control Loop

illustrated in Figure 108. Any one of these systems or any combination may be used to synthesize a closed loop attitude control system for any vehicle configuration.

The component weight and power requirements shown in Figure 108 are for the complete 3-axis control using the particular command mode.

C. CONTROL SYSTEM SELECTION BY SIMULATION

The attitude control characteristics of the vehicle configurations presented in this report were selected on the basis of the general considerations discussed in Section II. The selection of a satisfactory control system for any given vehicle is determined by the specific mission requirements, and usually involves a compromise between control system complexity and vehicle performance. The necessary determination of the improvement in control system performance resulting from each level of increased complexity, and the resultant effect on overall mission performance, can best be accomplished by means of a piloted simulation. Some aspects of the control system which may fruitfully be examined in this way are discussed below.

1. Control Power

The effect on mission performance of providing various levels of control power may be studied. Although very high values can be handled by means of automatic control loops, the lowest adequate value should be selected. This will minimize weight of the moment producing devices, provide finer control resolution and more desirable limit-cycle characteristics, and improve the ability of the pilot to control the vehicle in an acceleration control mode. Reduction of the available control moment to barely exceed that required to counter the maximum anticipated disturbance moments, however, will result in highly asymmetrical control power when such disturbances exist. Under these conditions, the available control power in the direction opposing the disturbance may be inadequate, even if it is higher than the minimum level required in the absence of disturbances. The tolerable degree of control power asymmetry for various control tasks may be determined by simulation.

2. Moment Trim System

The need for a separate trim system to handle large sustained disturbances, thereby permitting the basic attitude controller to be limited to lower values of control power (for the reasons discussed above) may be explored. The relative merits of manual and automatic trim, and the maximum allowable response rate of the latter, may also be determined.

3. Control Modes

The relative merits of acceleration command, rate command/rate hold, rate command/position hold and position command attitude control systems for various missions and mission segments may be explored.

4. Coupling Effects

The effect of coupling of attitude accelerations into translational or cross-axis attitude accelerations on control precision and mission performance may be studied. In particular, an examination of the effect on boost cutoff velocity, where cutoff is initiated on the basis of time alone and of the thrust of separate pitch and roll reaction jets firing in noncouples may be useful.

5. Thrust Vector Misalignment

The degradation of guidance system accuracy resulting from the varying alignment of the thrust vector with respect to the vehicle axes, which will result from engine gimballing for c.g. control, may be determined by simulation. One solution to any problem of this nature consists of mounting the inertial sensors to the engine rather than to the frame. However, the need for this system complication, which may be particularly severe where a simple optical sight is used, should be explored.

D. SIMULATION OF THE BACK PACK PROPULSION DEVICE

Although, for the larger vehicles, it can be assumed that the pilot's body motions do not appreciably affect the overall moments of inertia and the c.g. location (and, hence, the vehicle's motion), it may be an over simplification to assume that the same is true for smaller one-man vehicles.

The problem of a flexible man in a small vehicle is exemplified by the case of a one-man propulsion device in the form of a back pack (shown in Figure 87) attached to the pilot in his pressurized suit with the required life support and means of communication. The pilot's ability to change the relative orientation of parts of his body depends very much on the type of pressure suit used for the particular space mission. Allowing for the pilot's flexibility will require consideration of more complex dynamic response and control characteristics of the unit. It may also necessitate that the possibility of controlling attitude by means of conscious or reflex body motions be accounted for.

The performance evaluation of a back pack propulsion device should begin with an examination of the pilot's ability to move his legs, arms, shoulders, and body trunk since current extravehicular suits are restrictive of body movement. In this case, the evaluation (simulation) of the vehicle can be carried out along the lines valid for large rigid vehicles.

If, on the other hand, the particular choice of equipment permits limited body motion, then the simulation should account for the increased complexity in the response characteristics as well as the added control possibilities.

Depending on the assumptions made with regard to the pilot's influence on the overall vehicle response the simulation of a man-back pack unit may be conducted in one of four ways, ranging from a rigid or segmented body approach to the operation of a realistic unit in a simulated lunar gravity field.

Simulation of the rigid or segmented body approach would require an analog setup consisting of the analog computer, the displays and a control fixture. Care should be taken in programming to include all the computations necessary for the simulation. The basic components of this simulation would include the body dynamics and cross coupling effects represented by the equations of motion for either a rigid or segmented body, the display generation, the fuel consumption and the instantaneous c.g. and inertia variations.

The display console may be instrumental to display the fuel remaining and the attitude of the systems in pitch, yaw and roll by individual instruments, by a single indicator which displays all three functions or by a contact analog type of display. A more realistic presentation could be made using a wide angle display of the lunar surface which is presented to the pilot. From such a display, a large portion of the important visual cues can be presented.

The controllers should be mechanized to represent those installed on the actual system. For simulation of the back pack, an arrangement which duplicates the thruster configuration and deflection angles of the thrusters should be employed, and the operator preferably in a pressurized suit should be allowed to provide the commands. In this respect, the operators response, restrained by the pressurized suit can be realistically represented as they are related to the maximum deflections and attainable input rates to the gimballed thrusters. This arrangement may be employed to conduct the first three simulations described below:

(1) The rigid body approach is the simplest solution to the simulation problem. This approach is valid for relatively rigid spacesuits or for a flight technique requiring the pilot to hold himself rigid during flight. The pilot then flies this unit like he would any larger rigid vehicle by means of thruster gimballing, throttling and/or reaction jet operation.

(2) A relatively simple modification of the first simulation method consists of substituting a two-body system consisting of a large mass to represent the pilot's upper body and the back pack, a spring and dashpot to represent the hip joints, and a smaller mass to represent the pilot's lower body, for the single mass of the rigid unit. A man-back pack simulation of this kind will only account for the dynamic response characteristics of the flexible man, but it does not represent any voluntary body motion or reflex action of the pilot. Control and stabilization means for this unit are limited to those of the rigid, single mass unit. This simulation model may be further extended to consider smaller units of mass such as individual arms and legs, and to account for the additional degrees of freedom.

(3) A considerable improvement in the fidelity of the simulation may be obtained by extending the simple one- or two-body simulation to include an artificial stabilizing element to represent the pilot's intentional or unintentional body

motions which tend to aid in the stability of the actual vehicle, but are not otherwise included in the simulation. This form of simulation is shown in Figure 109.

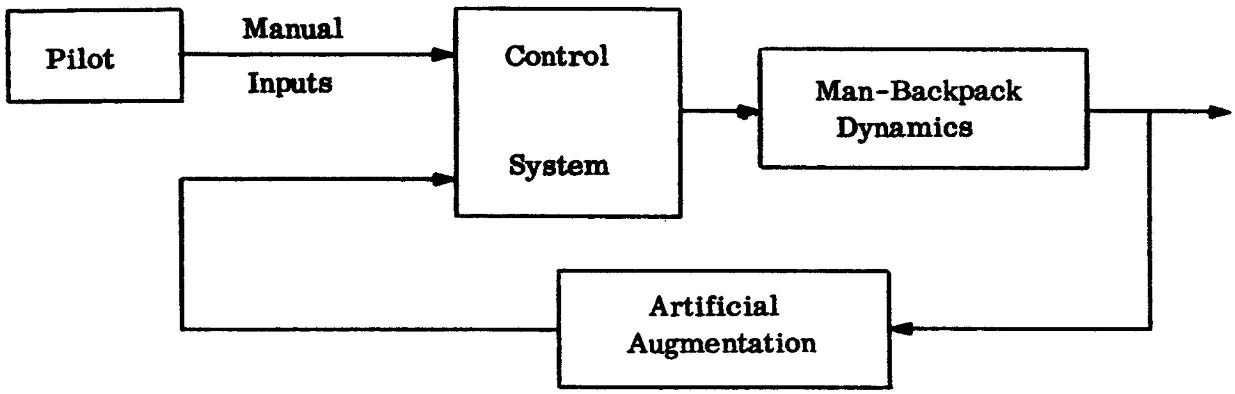


Figure 109. Back Pack Simulation Block Diagram

This approach has been used in simulation studies at the Bell Aerosystems Company with some success. The arrangement of this simulation is similar to that previously discussed with the exception of the artificial augmentation block shown in Figure 109. The stabilizing element was developed by simulating the Bell Small Rocket lift device and adjusting the parameters until experienced pilots indicated that the simulation exhibited about the same characteristic as the actual flight unit.

(4) The evaluation of a man-back pack unit may be approached by attempting to physically simulate the flight situation. This differs basically from the three simulation approaches described previously. In this case, the environment in which the actual unit will operate (lunar surface or zero g conditions), would be simulated for limited periods of time and a system representative of the actual unit would be operated in this simulated environment.

Simulation of the environment would require a mechanical setup: one possibility is a servomotor driven, gimballed rig on an overhead crane unit as illustrated in Figure 110. The representative system would include the pilot in his pressurized suit, life support equipment, tanks and thrusters.

The advantage of this approach to the simulation of the man-back pack unit would be the inclusion of all the man's contribution to the overall system performance - dynamic response, reflex action, and intentional body motions for control and stabilization of the unit.

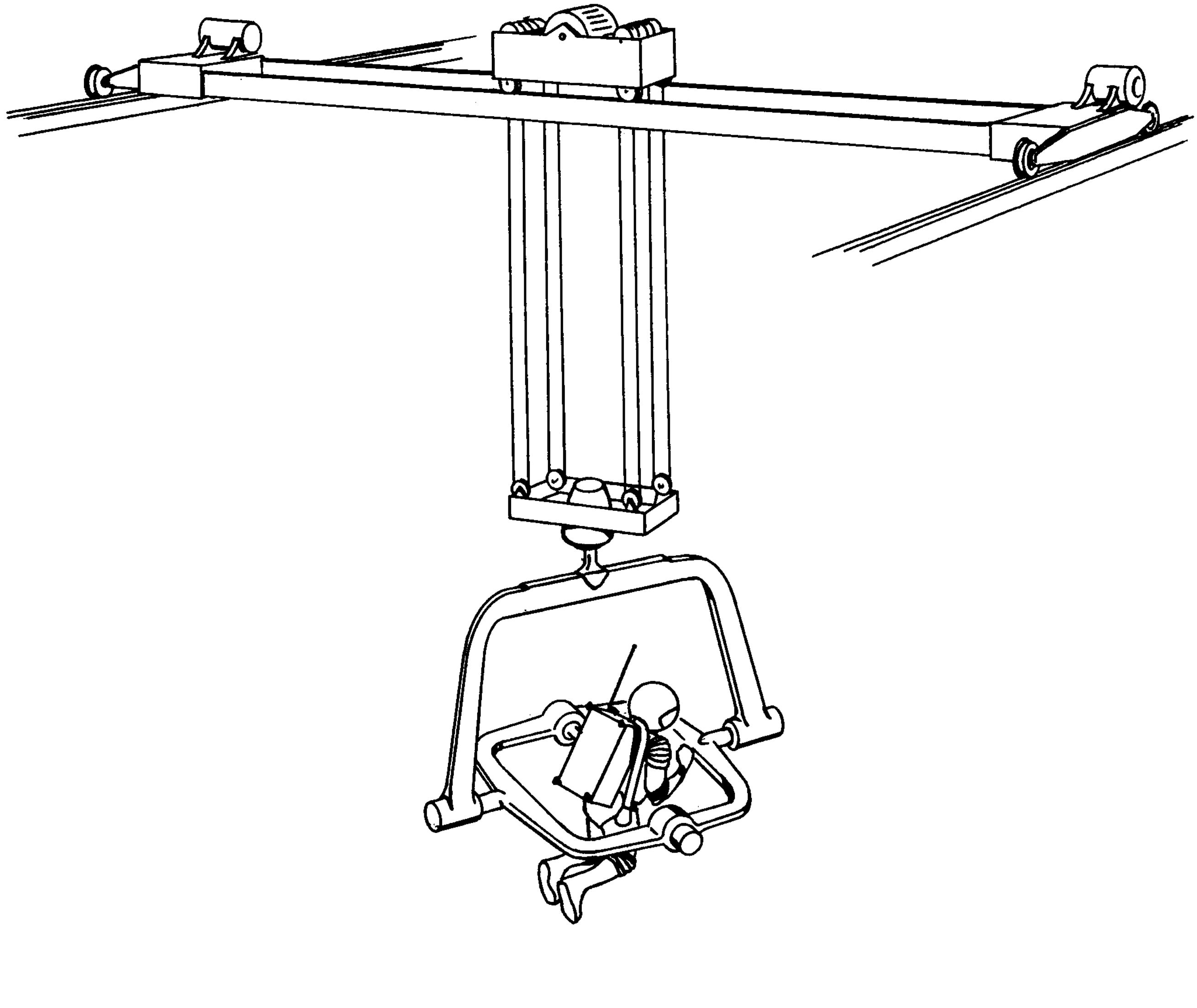


Figure 110. Overhead Crane Arrangement

The propulsion system used to simulate the back pack devices may use either hydrogen peroxide or gaseous N_2 as propellants. Although the hydrogen peroxide unit can provide longer periods of operation and more representative dynamic properties, a unit operating on gaseous nitrogen may be employed to produce limited periods of operation if the environment is such that operation of a hydrogen peroxide unit is not permissible.

The results of a preliminary investigation of a self contained gaseous nitrogen unit to be used in the simulation studies indicate that storage volume restrictions ($\sim 2\ ft^3$ at 3000 psia storage pressure) limit the operational capability of the simulation duration to approximately 6 seconds at maximum thrust ($F \approx$ 200 lb and I_{sp} at sea level = 40 sec) or approximately 20 seconds at hover thrust conditions. It is also evident that the weight distribution, the moments of inertia and c.g. excursions of such a unit would not be representative of the actual back pack which is being simulated because the high pressure nitrogen tanks differ both in weight and size from the actual tanks. It is therefore recommended that the model used in simulation be constructed to represent the dynamic characteristics of the actual unit, and a ground based nitrogen ($\sim$ 200 psi) supply be employed to provide various nitrogen to the thrust chambers. The duration of each simulation run thus obtained is not dependent on the amount of gas that can be loaded on the back pack but rather on the capacity of available ground based equipment which can considerably extend the available simulation period.

Another approach to the physical simulation for the man-back pack unit may result from the use of the Lunar Walking Simulator, which has been developed and is currently used at Langley to study the problems of walking on the moon (Reference 11). The model used in this simulation would be either the self contained nitrogen powered unit or the unit obtaining nitrogen supply from ground based equipment, depending on the duration of simulation desired. It is realized that the Lunar Walking Simulator restricts the simulation of the attitude to one degree of freedom. However, it allows complete freedom in the horizontal and vertical planes. It may therefore be used to conduct a realistic simulation of the pitch plane dynamics and to verify the envelope of permissible system weight and touch down residual velocity.

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